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POWER SUPPLIES AND TELEMETRY FOR AN INSTRUMENTED ARTIFICIAL SATELLITE

PART I. ORBITAL CONSIDERATIONS

BY E. C. WHITE

The present paper is concerned with the merits of various orbits which a satellite can attain round the Earth, the final choice being made only with due consideration to the amount of sunlight it can receive and the time it spends in darkness during each period. The latter is important because solar batteries are envisaged as a source of power supply.

The satellite is assumed to be spherical in shape and between two or three feet in diameter, and, because it depends on energy received from the Sun, it is essential that it should be eclipsed by the Earth for only short periods. (The first few satellites to be launched in connection with the International Geophysical Year programme will, however, be operated by chemical batteries.) The time of effective stay was arbitrarily fixed at 12 months and the range of orbits attainable in space are, of course, infinite in number. A circular orbit was decided upon, since, for a 12 months stay, careful orientation of an elliptical orbit for one period of the year would be cancelled out during another. This is because the position of the nodes of the ellipse are, ignoring nodal precession, fixed relative to the stars and not to the Earth-Sun system. No attempt is made in this paper to discuss the method of placing the satellite in its orbit or the feasibility of doing so. In order to reduce the problem to a relatively simple case, certain assumptions have been made, i.e.

(*a*) That the Earth's orbit round the Sun is circular. It is, in fact, an ellipse of present eccentricity ·017. The error in such an assumption is only of the order of $\pm$ 2 per cent. on radius and its effect on the configuration is quite negligible.

(*b*) The shadow cast by the Earth is assumed to be cylindrical. Fig. 1 will clarify this. The shadow cast by the Earth consists of two parts; the penumbra, which increases in intensity as the umbra is approached, and the umbra, which is of uniform intensity and approaches complete blackness. It is considered that the amount of sunlight which is refracted round the Earth by the atmosphere is unimportant from a power-supply point of view.

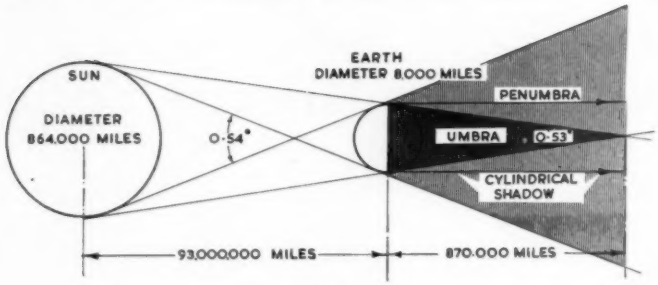

FIG. 1

The penumbra is a frustrum of a cone of 0·54° included angle and extends to infinity. The umbra is a cone of 0·53° included angle and extends for a distance of 870,000 miles. It will be seen that, in the region of the shadow in which we are interested, the difference between the boundary of the penumbra and the umbra is very small, and the line of the cylindrical shadow approximately bisects this distance.

In Fig. 2 the times the satellite spends in (*a*) the penumbra plus umbra, (*b*) the cylindrical shadow and (*c*) the umbra, are plotted for various heights of the satellite. The worst case of zero inclination to the ecliptic is taken. It can be seen that the curves are very close together and since the satellite will receive some sunlight even in the penumbra, the cylindrical-shadow assumption is a very close approximation.

The two factors which are of interest are:

(1) The amount of sunlight received by the satellite.

(2) The duration of total darkness in any one period.

These two factors are governed by the height of the satellite above the Earth's surface and the inclination of the satellite's orbit to the ecliptic. In order to deal with these two variables, they have been approached separately.

The worst case of inclination from a light-receiving stand-point is when the satellite's orbit is in the plane of the ecliptic, since, in this position, the satellite is eclipsed in part of its orbit at all times of the year, and furthermore, the eclipse time is constant for the whole year.

It will be seen later (Fig. 9) that at an altitude of 100 miles the satellite would be operating on stored energy for 37·66 minutes and this would decrease to 34·84 minutes at 800 miles. At 500 miles the time would be 35·14 minutes and it is thus obvious that, on the basis of time in shadow, the advantage in heights above 500 miles is small.

The time of stay also has an influence on the choice of height for the satellite. According to Dr. H. E. Newell, the life-time of such a satellite at a height of 100 miles is less than one hour, at a height of 200 miles it is about 15 days, and at a height of 300 miles almost one year. In the light of these factors an operating height of 600 miles was finally chosen as reasonable.

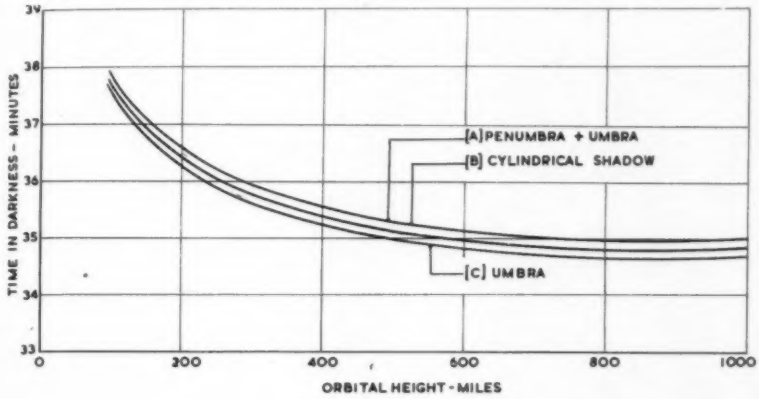

FIG. 2

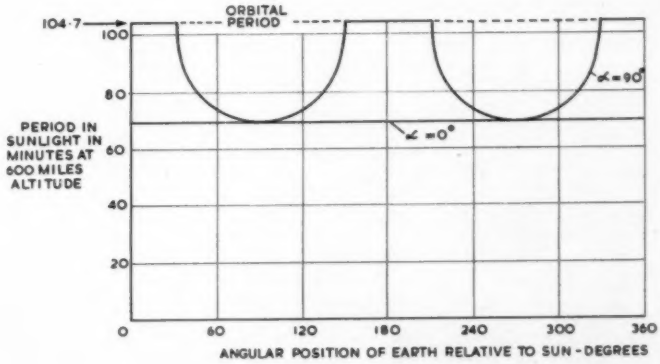

FIG. 3

The second factor influencing the time in darkness is the inclination of the satellite's orbit to the ecliptic. This will be referred to as α.

The maximum sunlight received occurs when $\alpha = 90°$ and a minimum when the inclination is zero. These two cases are plotted in Fig. 3 with period in sunlight and angular position of the Earth relative to the Sun in degrees (360° representing one year) as co-ordinates. The area under the curve is proportional to the total sunlight received by the satellite in one year. In the curve of $\alpha = 0$ the period in sunlight is constant throughout the year. This is to be expected since the shadow-satellite orbit configuration is independent of the time of the year. In the case of $\alpha = 90°$, the satellite is in full sunshine until the shadow of the Earth commences to eclipse the satellite, due to the Earth's motion round the sun. The rate of eclipsing is fairly rapid at first, but falls

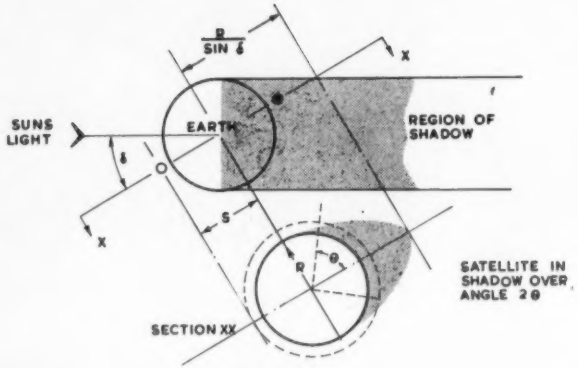

FIG. 4

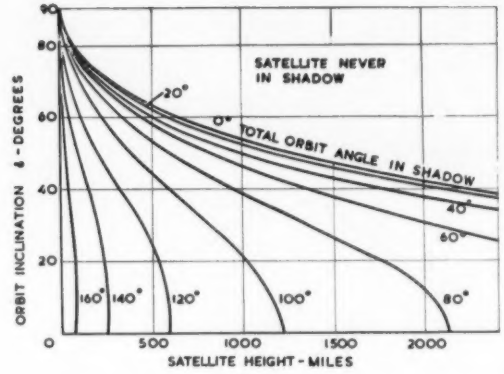

FIG. 5

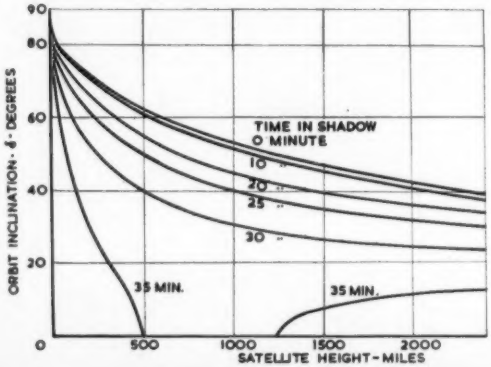

FIG. 6

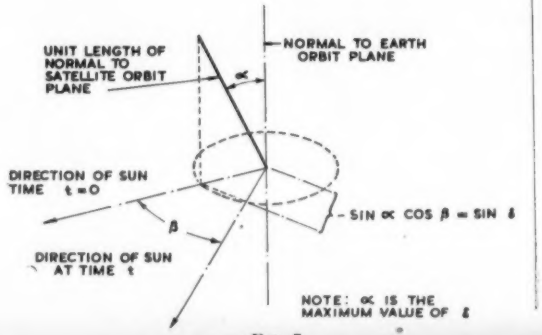
UNIT LENGTH OF NORMAL TO SATELLITE ORBIT PLANE
NORMAL TO EARTH ORBIT PLANE
α
DIRECTION OF SUN TIME t = 0
SIN α COS β = SIN δ
β
DIRECTION OF SUN AT TIME t
NOTE: α IS THE MAXIMUM VALUE OF δ

FIG. 7

off until a maximum of eclipse time is reached. This maximum is the same as that for $\alpha = 0°$. The amount of eclipse time then slowly decreases, the rate increasing as full sunlight is approached, the curve being repeated every half-cycle.

APPENDIX

In Fig. 4 the Earth is viewed in the plane of the satellite's orbit, showing the Earth's shadow extending to the right as a cylinder with radius R equal to that of the Earth. The satellite's orbital radius is S and the angle between the orbit plane and the direction of the Sun is δ. The diagram is a section made in the plane of the orbit and shows the shadowed area enclosed by an elliptical boundary. As the satellite crosses this boundary it enters or leaves the Earth's shadow.

The polar equation of an ellipse may be written

$$\frac{1}{r^2} = \frac{\sin^2\theta}{a^2} + \frac{\cos^2\theta}{b^2}$$

where a and b are the semi-axes. For the shadow boundary, the semi-axes are R and $\frac{R}{\sin\delta}$ and we wish to find θ when $r = S$. Making these substitutions

$$\frac{1}{S^2} = \frac{\sin^2\theta}{R^2} + \frac{\cos^2\theta\sin^2\delta}{R^2}$$

$$\frac{R^2}{S^2} = \sin^2\theta + \cos^2\theta\sin^2\delta$$

from which

$$\cos\theta = \frac{1}{\cos\delta}\sqrt{1 - \frac{R^2}{S^2}}$$

This relation SNP, has been used to plot the curves in Fig. 5 which gives the angle through which the satellite travels in the Earth's shadow for any inclination of the satellite orbit and for heights of up to 2,400 miles.

This information has been combined with the angular speed of the satellite to give the curves in Fig. 6, which shows the time the satellite stays in the shadow on each occasion.

In both cases the inclination δ of the orbit to the Sun must be known before the charts are of use. If α is the inclination of the satellite orbit to that of the Earth, then on two occasions each year, six months apart, the inclination of the satellites orbit to the Sun is equal to this, i.e., $\delta = \alpha$. Fig. 7 helps us to

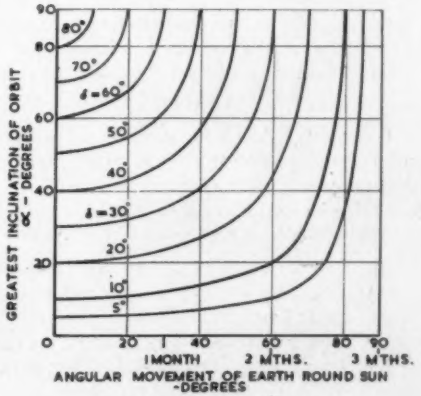

FIG. 8

determine the value of δ at other times. In this diagram β is the angle through which the Earth has travelled in its orbit since δ and α were equal, and the three quantities are connected by

$$\sin \delta = \sin \alpha \cos \beta$$

Fig. 8 is a chart which makes it possible for δ to be determined for values of α and β up to 90°. A simple manipulation of the chart allows any values of α and β to be used.

PART II. INSTRUMENTATION AND TELEMETRY

By J. Foley, B.Sc. (Eng.) G.I.Mech.E., A.F.R.Ae.S.

The satellite is best regarded as a means of extending current work on the exploration of the upper regions of the atmosphere. The measurements required bear a close relation to those already being made, the major difficulties arising in trying to get hold of the measured data. Every rocket has to return after, at most, a few minutes flight, and photographic and other records can be recovered by suitably protecting them and using parachutes to retard their descent. These methods are not applicable to the satellite, for information must be recorded over periods of days or weeks and recovery is almost certain to be frustrated by aerodynamic heating.

This indicates that a radio link is essential, and instrument readings must be converted into signals which can be decoded reliably after passing through a transmitter and receiver and several hundred miles of space. This has already been done for investigations using high altitude rockets, and interesting descriptions of the telemetry equipment used have been published, but we cannot adopt this equipment for the satellite without considerable revision for the following reasons.

(*a*) The power demand is considerable; several pounds weight of batteries are needed for a rocket flight of 15 minutes.
(*b*) Signals are sent continuously to the ground station, which needs to be in operation for the period of flight only.
(*c*) Greater consideration must be given to the life expected from components, for example by conservative rating, or by allowing rest periods.

Our method has been to make an assumption about the size of the satellite and the equipment it should carry, and then to devise the best means for supplying the power required by the instruments and transmitting equipment to send information back to Earth.

The subjects of our investigations are

(*a*) Cosmic rays of different energy levels.
(*b*) The intensity of the solar radiation in the X-ray and ultraviolet.
(*c*) Solar particles.
(*d*) The number and size of micro-meteorites.
(e) Earth's reflection of solar energy, both total and local.
(*f*) Magnetic field measurements.
(*g*) Temperature measurements on the satellite's surface.

Cosmic rays are severely modified by the atmosphere and are deflected by the magnetic fields of the Earth and possibly also of the Sun. They are detected by Geiger-Muller tubes, which are familiar from their widespread application to the measurement of radioactivity. An array of such tubes can detect the direction of arrival of individual particles, and Dr. Singer has proposed the use of the Earth's magnetic field and a circumpolar orbit as a way of discovering their energy distribution.

The radiations and particles which originate in the Sun can also be detected by G.-M. type equipment. Photon counters are quoted by Dr. Singer as being available for use in the X-ray and Ultraviolet regions of the Sun's spectrum, and we shall assume that their power requirements will be similar to those used for cosmic ray detection. Measurements of the strength of the Sun's radiation, which is absorbed by the atmosphere, have been made with a rocket-borne grating spectrograph, but this work relied on the recovery of photographic records and the equipment for orienting such a spectrograph is more than can be installed on this satellite. Another way of making measurements in narrow bands of the spectrum is to use a motor to interpose a series of filters between the Sun and a photocell, a technique used in the Aerobee in determining the nature of the light from the sky.

Microphones mounted in the outer skin of rockets have been used for the detection and counting of micro-meteorites, and may be adapted directly for use in the satellites.

The photo-cell equipment mentioned in connection with Aerobee may also be developed to make observations of the Earth's reflected light, but difficulty will arise in scanning the Earth's surface for more than a short period without mechanical complexity. It will also be necessary to devise some means of ensuring that the detector is not damaged by receiving the direct light of the Sun.

Magnetometer measurements of the magnetic field strength at high altitudes have been made with rocket-borne equipment, but the available information does not allow an estimate of the power requirements to be made, we are forced to eliminate it from further consideration.

We are also interested in the temperatures experienced by the surface of the satellite in space, and in the heating caused by its high speed of movement through the upper layers of the atmosphere as it falls back to the Earth. Foil resistance thermometers were used for similar purposes in one of the Viking experiments and may be used again without serious change.

The instruments we have now considered do not all give an output which can be applied directly to the modulator and transmitter, and some change of form must be imposed upon their output signals.

Instruments giving a series of pulses which indicate individual events, for example a Geiger-Muller counter indicating the arrival of a cosmic ray particle, or the micrometeorite detecting microphone, may be connected directly through an amplifier, but even in this case, as is shown later, it may be advantageous to change the form of the signal.

Photocell equipment observing the intensity of slowly varying illumination will give an output voltage level proportional to the light intensity. This voltage is compared with some standard source carried within the satellite by

TABLE I

ESTIMATED WEIGHTS AND POWER REQUIREMENTS OF EQUIPMENT

Instrument	Auxiliary equipment	Current mA.	Voltage	Power W.	Approx. weight lb.	Duty %
Counter tube		0·05	700	0·35	0·15	100
	Quench amplifier, first	4	200	0·8	0·25	100
	dividing stage ..	50	4·5	0·225		
Microphone ..		4	200	0·8	0·25	100
	Amplifier and limiter ..	50	4·5	0·225		
Thermometer		10	50	0·5		
	Voltage comparison ..	4	200	0·8	0·25	6
		50	4·5	0·225		
Photocell ..		0·2	1000	0·2	0·25	6
	Pre-amplifier	2	200	0·4	0·15	6
		50	4·5	0·225		
	Binary pulse counter ..	34	4·5	0·153	0·5	100
	Transmitter	20	200	4	0·75	6
		600	6·5	4		
	Motor for switch or filters			6	0·5	6
	Receiver	12	200	2·4	0·35	100
		100	4·5	0·45		
	Modulator			1	1·0	6
	Power supply converters	30 % Supplied power			1/7 lb. per W.	

a voltage comparison circuit which produces a pair of pulses separated by an interval from which the voltage may be known. A photocell used for scanning the Earth's surface, or the spectrum produced by a grating, gives an output varying with time. This may be used to modulate a C.W. transmitter, but calibration becomes difficult unless a source of known intensity, such as a part of the Sun's spectrum measurable from the Earth's surface, is included in the signal. A voltage output will also be given by the resistance thermometer.

Now we should consider the operation of the equipment as a whole. One of the big advances made possible by the satellite is the ability to get a continuous flow of information for a long period. The advantages of periodic transmission of data; which are economy in power demand, reduction in the number of ground receiving stations and improved life of the more heavily loaded components; must be paid for by the necessity of a data-storage system. It is not easy to devise this for slowly varying quantities, but, by their very nature, these need such a system less. Periodic measurement may be sufficient if the satellite can be made to answer requests for information by including a receiver operated switch.

Cosmic ray and particle counting instruments give a series of pulses and our main interest is in the number arriving in a given time. This type of information can be stored in a set of binary counting stages, each of which gives an output pulse for every second input pulse. The set can be made to reveal the number of pulses it has received by the currents taken by the successive stages, which may be read as a binary number. From figures of the total cosmic ray flux taken in American experiments the number of stages required for the greatest

TABLE II

Component	Continuous transmission		Periodic transmission	
	Mean power watts.	Weight lb.	Mean power watts.	Weight lb.
Two Geiger counter tubes ..	2·75	0·8	2·75	0·8
Resistance thermometer ..	1·53	0·25	1·53	0·25
Photocell	0·82	0·35	0·82	0·35
Transmitter, duplicated ..	8	1·5	0·48	1·5
Switch motor	6	1	0·36	1
Modulator	1	1	0·06	1
Data storage	—	—	0·31	1·6
Receiver	—	—	2·85	0·35
Power converters	3	2	1·65	2·5
Totals	23·1	6·9	10·8	9·4

possible number of counts in one orbital cycle of 90 minutes is 17. The weight of the complete counter would be about 8 oz. if transistors are used for all but the first stage (which should be filamentary valves for the sake of their frequency response) and the power requirement about one fifth of a watt.

The number of pulses must be "read out" by an electronic or motor driven switch, and the resultant series of voltages transmitted to be read on the ground. This will happen when the satellite's receiver is given the appropriate signal from the ground station.

At this stage a word on the choice of components seems appropriate. The transmitter has to deliver about 2 watts at a frequency exceeding 50 Mc./s and will need an input of about 4 watts anode supply and the same for heaters, since heater-cathode valves are necessary. Filamentary cathode valves will be used for amplifier stages where a good frequency response is needed, and for voltage-comparison circuits. Other stages of amplification, pulse modulation, and counting and switching circuits will use transistors. These are far lighter and more economical to run than other types of component, but are subject to variation of characteristics with temperatures, and have limited frequency response and power dissipation. They are well suited to modulators and power supply converters since their losses may be arranged to be low, and can handle powers of three to four times their nominal dissipation. Two-watt transistors are a well-established laboratory component.

The least reliable of the three types is likely to be the transmitter valve, 1 per cent. replacement per hundred hours being reported as typical in one airborne equipment. This means that it is worth using a duplicate satellite transmitter, the pair to be used alternately. Receiver-type valves are two or three times more reliable, so the receiver may not need duplication. Transistors in suitably designed circuits are extremely reliable, at least twice as reliable as receiver valves, though it is a little early to expect full life information to be available.

Table I lists the equipment and gives estimated weights and power requirements. Table II shows a possible selection of instruments for a satellite required to send a continuous stream of information, and shows the changes

and economics made by adopting some means of storing information and transmitting it periodically.

PART III. AVAILABILITY OF POWER

By R. G. Wilkins

One of the most important considerations in dealing with the power unit for the instrumentation of any airborne or rocket-borne vehicle is to keep the weight as low as possible. This is particularly important in the case of an artificial satellite of the type we are considering, as the propulsion technique is virtually at the limit of its possibilities.

Such a power supply can be obtained from two types of source. In the first place batteries, either of primary cells, e.g., ordinary "dry batteries"—or of accumulator cells, can be used. The latter especially can be obtained in light-weight types. The second possibility is to take the power from some external source, obviously the Sun in the case of an Earth satellite, via. some form of converter.

The first possibility is practicable only for short times of stay of up to a few hours duration. It is inherent in the nature of the two forms of battery mentioned, that primary cells are more efficient with respect to power/weight ratio when delivering comparatively low currents and high voltages, as required for the high tension supply for electronic valve circuits.

The accumulator, because of its much lower internal resistance, is more suited for low voltage/high current applications. For instrumentation employing traditional electronic valves, dry batteries and accumulators have been used together. For very large payloads, such as in the German A-4 rocket, accumulators have been used in conjunction with a rotary converter for the high voltage supply.

It is expected that for satellite vehicles transistors will be used wherever possible. For these a supply of the order of 4 to 5 watts is required at currents of the order of $\frac{1}{2}$ to 5 mA. for each transistor. For the more complicated systems having upwards of 100 transistors the current can amount to $\frac{1}{2}$ amp. or so. Some electronic valves will be required for the transmitter telemetering the instrument recordings to ground and for a responder receiver if required. These can be supplied from the same source, as light and efficient inverters can be constructed using transistors. For much longer times the weight of batteries becomes prohibitively heavy, and the possibility of using the Sun's energy must be considered.

The Sun's radiation can be considered for our purposes, as that emitted by a so-called black-body radiator at a temperature of 5670° Absolute. The resulting energy at the Earth is about 2·04 calories/sq. cm./min. at the top of the atmosphere. This is equivalent to 1,420 watts per sq. metre of collecting area. There is a periodic variation of about 5 to 6 per cent. in this figure.

The "classical" method of utilising this energy, using a mirror concentrating the sun's rays on to a boiler generating steam or mercury vapour for use in a more or less conventional turbo generator set, has been considered by several authors.[1,2] Stuhlinger[3] has quoted a figure of 20·5 watts/lb. for a large

200 kW. plant of this type for use in an ion rocket. For very small generators this figure would become very much worse. A mirror is essential in a plant of this type in order to increase the boiler temperature sufficiently so as to attain a reasonable efficiency. It does not appear at all practical to employ a mirror in a small satellite of the type we are considering owing to its weight and practical difficulties in erection.

It may be mentioned here that the efficiency as normally defined, i.e., as the ratio of power output available to power put in, does not directly affect the weight of the power unit, but it does determine the size of radiation collector to be employed for a given power requirement and this may in turn affect the weight.

There are two methods by which a direct conversion of radiant into electrical energy can be achieved, the thermopile and the semiconductor solar converter.

The thermopile consists of a large number of conductors of two differing metals joined alternately in series, alternate junctions being heated and the remainder cooled. The voltages produced per pair of junctions is quite small, at the most about 300 μV. per degree temperature difference between hot and cold junctions. A figure less than 100 μV./degree is more usual.

It can be shown[4] that the efficiency of a thermojunction depends on its thermoelectric power (the voltage per degree temperature difference), the electrical conductivity and the thermal conductivity. Suitable materials would have a high electrical conductivity, low thermal conductivity and high thermoelectric power.

Materials having the highest thermoelectric power so far obtained are alloys of Antimony, Bismuth and Zinc. Unfortunately they also have low electrical conductivities. They also have low melting points (200° to 300° C.), but for our application this does not matter as the maximum temperature reached for blackened bodies at the Earth's distance from the Sun is only about 116° C[5]. The efficiencies obtainable are very low—mostly less than 1 per cent. but an efficiency of 3½ per cent.[4] has been obtained using a couple consisting of an alloy of 21 per cent. Bismuth and 9 per cent. Antimony against an alloy of 35 per cent. Zinc, plus 65 per cent. Antimony.

Small amounts of Tin, Silver and Bismuth were added to the latter to improve its rather low electrical conductivity and to reduce brittleness. The temperature range employed was 20° to 200° C. In spite of the lower hot junction temperature which we will have available, the cold junction temperature can be made much lower (about − 240° C.)[5] simply by exposing the cold junctions to "Space" away from the Sun, so that one may expect an improvement in efficiency due to this, to perhaps 7 per cent., the efficiency being roughly proportional to the temperature difference.

A thermocouple of lead sulphide and the already mentioned Zinc/Antimony alloy has given an efficiency of 7 per cent. but lead sulphide being a semiconductor the effect does not appear to be purely thermoelectric. It is not known what improvement there would be under the temperature conditions attainable in the satellite.

Stuhlinger[3] has derived a figure of 2·3 watts/lb. for the power/weight ratio of a couple of constantin (60 per cent. Copper + 40 per cent. Nickel) and the Zinc/Antimony alloy already mentioned. The efficiency of this couple is

lower, however, only 2½ per cent. and a value of 5·5 watts per lb. is reasonable for the Bi/Sb-Zn/Sb couple under space conditions.

The greatest disadvantage of using a thermopile is that the unit must always be correctly orientated with respect to the Sun. Some form of servo will be required for this and the weight will be considerably increased. It may be borne in mind however that for some types of instrumentation, as for example when taking solar data, an orientated platform may be required in any case.

The second type of radiation—electrical energy converter is the Silicon solar battery developed by Bell Telephones Corp. in America, primarily for supplying power for remote rural telephone systems.

In an ordinary metallic conductor, e.g., copper or iron the ability to conduct electricity can be explained by assuming that the outer or valency electrons of the atoms, i.e., those which normally take part in chemical reactions, are in a free state existing as a type of gas in the spaces in the crystal lattice formed by the symmetrically arranged atoms. On applying a potential difference between any two points in the metal there will be a drift of the negatively charged electrons towards the positive pole. This constitutes the electric current.

In non-metallic conductors, particularly carbon, silicon and germanium, the valency electrons are not free but take part in the bonding of the atoms in the lattice. In order to cause a current to flow sufficient energy must be applied to free the electrons from their positions. Such materials are known as semiconductors.

A semiconductor such as silicon can be modified in two ways by adding small amounts of impurities. Silicon has a valency of 4, that is there are 4 outer electrons in its structure available for chemical reactions. If small amounts of an element having a valency of 5 such as Arsenic or Antimony are added then for every atom of impurity there will be one free electron available for conduction. It is assumed that the amount of impurity added is sufficiently small not to modify the lattice structure. The silicon is then designated *n*-type (one extra electron ≡ one (negative) charge). If however the impurity has a valency of 3 (e.g., Boron or Aluminium) there will be one electron missing for each impurity atom. This is technically known as a "hole." Conduction is possible by migration of an electron into a hole. The hole behaves as if it were a positive electron, migrating in the opposite direction to the electron flow. This is known as *p*(positive)-type silicon. The hole is normally filled with a stray electron.

If a junction is made up of the two modified types of silicon and the *p*-type silicon exposed to radiation, the incident photons, if of sufficient energy, create pairs of electrons and holes. Because of the close proximity of the *p*-*n* boundary the electrons and holes are separated, the electrons migrating across to the electron excess *n*-type material. An e.m.f. is thus set up and a current can be taken from the system.

The e.m.f. of a single unit is about ½ volt[6] under open circuit conditions but drops as the load is decreased. It is found that the voltage at which the efficiency is greatest is independent of the intensity of radiation. This means that the most suitable load is one working under constant voltage conditions. A single unit can be constructed from a slab of *n*-type silicon covered by a thin layer of *p*-type silicon on which the radiation falls.

Only part of the Sun's radiation is of use in creating "electron" hole pairs. All radiation having energy less than that required to eject an electron is useless. The threshold energy is 1·02 ev., corresponding to a wavelength of 1·2. Further all photons of energy greater than this have the same effect, creating only one electron-hole pair up to energies of 2·04 ev. when they may then eject two, but the incident radiation intensity is falling off rapidly in this region and comparatively little radiation has this energy. The result is that there is a maximum possible efficiency of 23 per cent. This must be further reduced by allowing for reflection of some of the radiation at the surface of the *p*-layer. In practice a value of 17 per cent. has been obtained.

Although the silicon generator appears much superior to the thermopile it has one disadvantage. This is that it may be more susceptible to damage from cosmic rays which would take the form of an eventual reduction of the impurity centres. This effect becomes progressively more severe the longer the time of stay of the satellite but may be negligible for times of a few days duration.

We will now consider what power can be obtained from a battery of these silicon cells and will assume the satellite has the form of a sphere 30 in. diameter, a size which has been put forward elsewhere. The silicon elements are arranged in eight segments over the whole sphere, separated by strips 4 cm. wide which will be a part of the main frame of the sphere. The cells themselves can be made in any size, the largest practicable being about 6 cm. × 1½ cm.[6] Allowing for the separations between octants and ·05 in. between each cell a sensitive surface of 77 per cent. of the spherical surface is available. The projected surface "seen" by the Sun is 0·707 sq. in. 77 per cent. of this is ·542 sq. in. and the radiation falling on this area amounts to 500 watts. If we assume the efficiency of conversion to be 11 per cent. then the power available is 55 watts.

Taking the silicon cells of the type used by Bell Telephones which were 5·7 cm. × 1·2 cm. × 0·1 cm. thick, we obtain a power/weight ratio for the cells alone of 33·7 watts per lb. Allowing for a light weight backing, connections, etc. it seems reasonable to halve this figure, say 17 watts/lb. This gives a weight of 3·52 lb. for the 55-watt generator.

Half of the cells will of course be facing away from the Sun and therefore useless. In order to prevent the dissipation of electrical energy in the internal resistance of these cells which would still further lower the efficiency it is desirable to arrange some sort of switching to short out those cells which are out of action. The simplest way would be to include a bimetal temperature switch with every few wafers to short them out when no radiation falls on them. Four switches per octant would be sufficient. A temperature range of nearly 400° will occur if the strips are blackened so that only a comparatively light and insensitive switch would be needed.

Because, depending on the orbit chosen, the satellite will be in the Earth's shadow for part of the time, some comparatively small auxiliary power supply will be needed. This is most conveniently provided by arranging that the solar converters can continuously charge a battery of accumulator cells. The lightest accumulator cells available at present are the silver-zinc type having a capacity of 40 to 50 watt hours/lb. The weight will depend naturally on the period of darkness and the graph (Fig. 9) shows this as a function of satellite

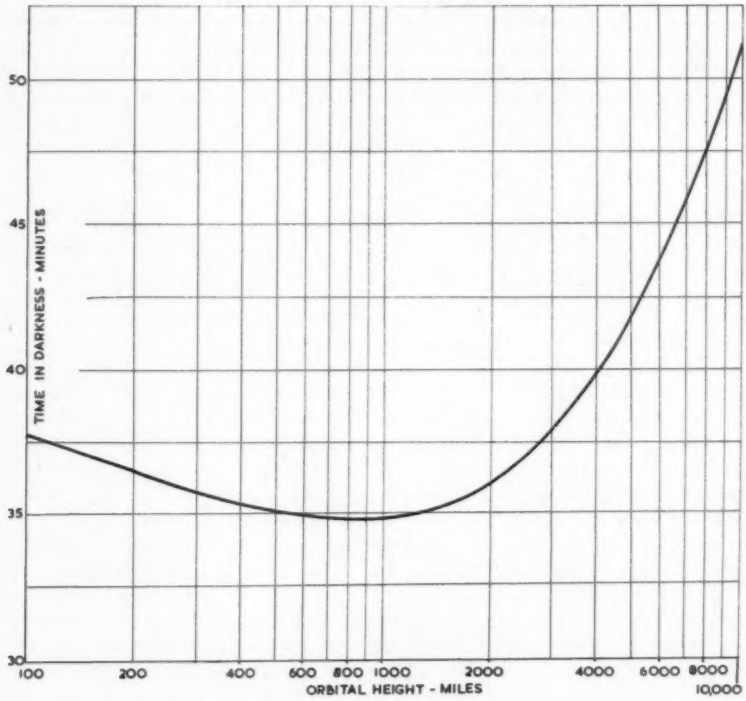

FIG. 9

height for the worst orbit, one in the plane of the ecliptic. It will be seen that for the orbit heights in which we are here interested the period of darkness does not vary over a very great range. If we assume a value of 0·64 hr. to cover most of our requirements, then for our 60 watt generator the watt hours required for the dark period is 38·4.

The weight of silver-zinc cells of this capacity will be about 0·77 lb. The total weight of the cells and converter is 4·29 lb. giving a power/weight/ratio of 14 watts/lb. The corresponding weight for thermojunctions, bearing in mind that an accumulator of similar size will be required will be 27·3 lb. giving a power/weight ratio of 2·22 watts/lb.

If accumulators alone are used then the greatest time of stay, if the power supply is to be no heavier than the silicon generator set, is 3·6 hr. A satellite having an instrumentation life of this time could complete at least one orbit at any height under about 3,000 miles.

To summarize, the weight factor and efficiencies of the possible forms of power supply have been collected together in Table III.

TABLE III

Type of supply	Efficiency %	Power/wt. watts./lb.	Advantages	Disadvantages
Turbo-electric generator	10	$\ll 20$	—	Mirror required. Power/weight ratio becomes rapidly worse for small sizes.
Thermopile alone .. .. with storage cells ..	$3\frac{1}{2}$–7 —	$2\frac{1}{2}$–6 2	Comparatively free from cosmic-ray damage.	Requires orientating.
Silicon solar generator alone .. .. with storage cells ..	12 —	17 14	No orientating required.	Risk of cosmic-ray damage.

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HEAT TRANSFER IN ROCKETS

By George P. Sutton* M.S., A.A.

In almost all designs and operations of rockets, considerable heat is transferred, and the phenomena of heat transmission often control the proper functioning of the units. This paper gives a general and brief discussion of some of the more significant applications of heat transfer theory in the field of rocketry.

The discussion is admittedly elementary and is intended for the students interested in rocket heat transfer problems.** The material is not new or not the result of any single research investigation, but rather a summary of fundamentals as they apply to various heat transfer problems in the field of rockets.

General Discussion

Heat is transferred from the hot reaction gases to the walls of the combustion device. In the general sense these devices include all types such as solid propellant combustion chambers and nozzles, liquid propellant gas generators and their hot gas piping, and injectors, nozzles, and chambers of liquid propellant thrust chambers.

Here these combustion devices will be divided into two categories: those that are *cooled* by liquid propellants and those that act as a heat sponge and are essentially *uncooled*. In cooled combustion devices the heat transfer usually reaches a thermal equilibrium condition. In an uncooled unit the operating time is usually short and the heat transfer is a transient phenomenon.

There are at least three accepted ways of cooling the walls of a thrust chamber with liquid propellant: *regenerative cooling* denotes the type in which the rocket chamber is cooled by circulating some or all of the liquid propellant (fuel or oxidizer) through a jacket (or coils) around the combustion device. The heat absorbed by the coolant is therefore not wasted but actually augments the heat content of the propellant prior to combustion. *Film cooling* implies that either the fuel or the oxidizer liquid is injected into the combustion devices in such a manner as to form a protective film of liquid, vapour, or cool gas adjacent to the walls. *Transpiration or sweat cooling* permits uniform, continuous injection of liquid over the entire surface of the wall to be cooled, by using a porous wall material, through which the propellant is fed.

Uncooled chambers are used in almost all solid propellant rockets and in some short duration liquid propellant rockets. Here the reduction of heat transfer to the wall by means of insulation and clever design usually permits a lowering of the chamber weights. Ceramic materials, special paints, and slow-burning organic materials bonded to the wall have been used here with success.

The selection of propellants with favourable heat transfer characteristics is another general topic. When liquid propellants are used to cool combustion

* This article is based largely on Chapter 5 of the second edition of the book "Rocket Propulsion Elements," which is to be published by John Wiley & Sons, Inc., New York, in the near future.

chambers, the significant, physical, and chemical properties must be considered in determining the ability of the propellants to withstand the high rate of heat flux and to absorb the heat. The cooling properties particularly desirable include:

(1) A high boiling point will permit a substantial temperature rise in the cooling jacket.
(2) A high specific heat, a high heat of vaporization and a high density of the liquid will increase the amount of heat absorbed.
(3) A favourable viscosity and conductivity will permit a favourable liquid film coefficient.
(4) Chemical stability, that is the resistance to change in chemical composition is very desirable, and
(5) The effect of impurities and manufacturing tolerance in the above properties and the variations with temperature should be small.

It must be understood that these are parameters that affect heat transfer only and the selection of the liquid propellant must be based on other factors in addition to that listed above.

Heat transfer is a significant for solid propellant units where it is desired to control the heating of the unreacted propellant grain while combustion occurs in the chamber.

Heat transfer problems are also important in determining the energy emitted from jet flames surrounding test stand structure. Other heat transfer problems in rocketry include aerodynamic heating of flying vehicles as it affects the environment of the rocket engine and the propellant tanks, and the heating of the working fluids in heat exchangers which are part of many rocket engines. These problems are separate from those listed under the heat transfer combustion devices and are not specifically discussed here.

General Steady State Heat Transfer Relations

For heat transfer conduction the general relation applies:

$$Q/A = -k\frac{dT}{dL} = -k\frac{\Delta T}{L} \qquad \ldots \qquad \ldots \qquad (1)$$

where Q is the heat transferred per unit time across a surface A, dT/dL is the temperature gradient with respect to thickness at the surface A, and k is the thermal conductivity expressed as the amount of heat transferred in B.Th.U. per unit time through 1 ft.^{2} of surface for 1° F. Temperature difference over a wall thickness of 1 ft. The negative sign indicates that temperature decreases as thickness increases.

The convective heat transfer through the chamber wall of a rocket can be treated as a series type, steady state heat transfer problem with a large temperature gradient across the gaseous film on the inside of the chamber wall, and a third temperature drop across the film of the moving cooling fluid. The problem is basically one of heat and mass transport associated with conduction through a wall. It is shown schematically in Fig. 1.

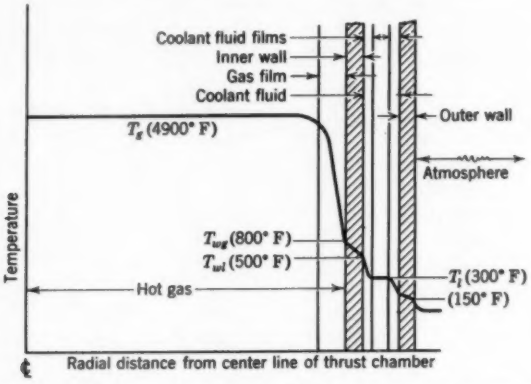

FIG. 1. Temperature diagram of liquid cooled thrust chamber.

The general steady state heat transfer equations can be expressed as follows:

$$q = h\,(T_g - T_l) = Q/A \quad .. \quad .. \quad .. \quad (2)$$

$$= \frac{1}{1/h_g + t_w/k + 1/h_l} \quad .. \quad .. \quad .. \quad (3)$$

$$= h_g\,(T_g - T_{wg}) \quad .. \quad .. \quad .. \quad .. \quad (4)$$

$$= (k/t_w)\,(T_{wg} - T_{wl}) \quad .. \quad .. \quad .. \quad (5)$$

$$= h_l\,(T_{wl} - T_l) \quad .. \quad .. \quad .. \quad .. \quad (6)$$

where q is heat transferred per unit area per unit time, T_g is the absolute chamber gas temperature, T_l is the absolute coolant liquid temperature, T_{wl} is the absolute wall temperature on the liquid side of the wall, T_{wg} is the absolute wall temperature on the gas side of the wall, h is the overall film coefficient, h_g is the gas film coefficient, h_l is the coolant liquid film coefficient, t_w is the thickness of chamber wall, and k is the conductivity of wall material. Any consistent set of units can be used in these equations.

Since the film coefficients, the gas and liquid coolant temperatures, the wall thickness, and the surface areas usually vary with the axial distance within a combustion chamber (assuming axial heat transfer symmetry), the total heat transfer per unit time Q can be found from:

$$Q = \int q\,dA = \pi \int Dq\,dL \quad .. \quad .. \quad .. \quad (7)$$

Since both q and D are complicated functions of L, the equation usually has to be solved by numerical methods. The arbitrary division of the rocket chamber into finite lengths and the assumption that q is given by equations 2 to 6 and remains constant over the length of each element will give an approximation of a solution.

TABLE I

Per cent. change in. film coefficient		q		
Gas film	Liquid film	B.Th.U./sec. in.^{2}	T_{wg} ° F.	T_{wl} ° F.
50	100	1·22	124	118
100	100	2·43	147	135
200	100	4·81	193	169
400	100	9·45	288	235
100	200	2·41	181	169
100	400	2·37	248	236
100	800	2·32	368	256
100	1,600	2·20	614 ·	603

The important quantities for controlling the heat transfer across a rocket chamber wall are the fluid film boundaries established by the combustion products on one side of the wall and the coolant flow on the other. The gas film coefficient largely determines the numerical value of the heat transfer rate. Known equations are based on experimental data and cannot be accurately applied to calculate this gas film coefficient, because rocket conditions differ from conditions existing in ordinary heat transfer problems.

Compared to conventional heat transfer problem the energy released per unit volume, the heat flux per unit area, the temperature and the surface velocity of the gases are of a different order of magnitude. Furthermore, the chemical composition and the physical properties are not very well known. The boundary layer effects in a non-steady flowing device such as a rocket combustion chamber are difficult to evaluate and equilibrium velocity profile conditions do not prevail. Diffusion effects, which become significant above 4,500° F., are usually neglected in rocket heat transfer calculations. The specific design conditions such as the injector design in liquid propellant rockets or the grain designs in solid propellant rockets have a profound influence on the gas film.

No simple theory exists now which covers all the important variables in the heat transfer of hot rocket gases to the wall. Until research provides accurate and adequate values of the gas film coefficient in rockets, the following limited method is recommended:

$$\frac{h_g d}{k} = 0{\cdot}023 \left(\frac{Dv\rho}{\mu g}\right)^{0.8} \left(\frac{\mu c_p g}{k}\right)^{0.33} \quad .. \quad .. \quad .. \quad (8)$$

where h_g is the film coefficient in B.Th.U./sec. R ft.^{2}, D is the diameter of the chamber in feet, v is the calculated average local gas velocity in ft./sec., k is the conductivity of the gas in B.Th.U./sec. ft.^{2} °R/ft., μ is the absolute gas viscosity in lb. sec./ft.^{2}, $c\rho$ is the specific heat of the gas in B.Th.U./degree/lb., and ρ is the gas density in lb./cu. ft.

As shown in a typical numerical example (Table I), the gas film coefficient h_g influences the overall heat transfer rate but has relatively small effect on the wall temperature. The exact opposite is true for variations in the liquid film

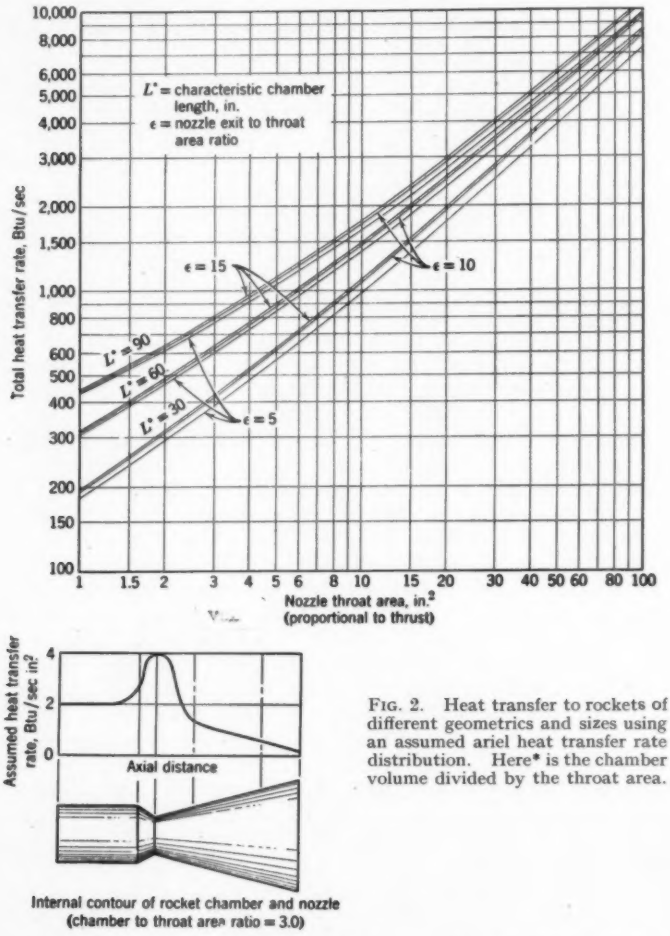

FIG. 2. Heat transfer to rockets of different geometrics and sizes using an assumed ariel heat transfer rate distribution. Here* is the chamber volume divided by the throat area.

coefficient; here changes in h_l (constant h_g) produce little change in the heat transfer rate but a substantial change in the wall temperature.

Rocket Thrust Chamber Heat Transfer

Heat is transmitted to all parts exposed to hot gases, such as injector faces, nozzle, and chamber walls. The heat transfer or heat transfer intensity, that is, heat transfer per unit area, varies within the rocket and is usually highest at, and immediately upstream of, the nozzle throat with local wall temperatures having the highest values in this region. Rocket thrust chambers with local

heat transfer rates of 0·1 to 10·0 B.Th.U./sec. in.^{2} have been successful. A typical heat transfer rate distribution is shown in Fig. 2. Only $\frac{1}{2}$ to 5 per cent. of the total energy generated in the gas is transferred as heat to the chamber walls. For a typical rocket of 10,000 lb. thrust the heat rejection to the wall may be between 700 and 3,000 B.Th.U./sec., depending on the exact conditions.

The amount of heat transferred by *conduction* from the chamber gas to the walls in a rocket thrust chamber is negligible. By far the largest part of the heat is transferred by means of *convection*. A part (5 to 25 per cent.) of the transferred heat is attributed to *radiation*. Radiation effects are more pronounced on large rocket thrust chambers than in small ones, because the mass of radiating substance is bigger in a large chamber; the radiation effects are treated at the end of this paper.

The amount of heat transfer increases as the exposed wall surfaces of the combustion chamber increases. For constant chamber pressure, the surface area of a combustion chamber generally increases with thrust. Usually the chamber wall surface increases less rapidly than the volume as the thrust level is raised. The cooling of combustion chambers by liquid propellants is thus generally easier in large thrust sizes, and the capacity of the coolant to absorb all the heat rejected by the hot gas is generally more critical in smaller sizes, because of the volume-surface relationship. The nozzle surface area of a combustion device is not subject to a simple surface-to-volume relationship and must be considered separately. As the nozzle exit areas become larger, the exposed nozzle wall surface subject to heat transfer increases.

Variations in chamber pressure seriously affect the heat transfer. In many cases the heat transfer rate per unit surface area has been found to be almost proportional to the chamber pressure; theoretically the gas film coefficient is proportional to the 0·8 power of the gas density. The increase of heat transfer with chamber pressure often imposes design limits on the maximum practical chamber pressure for both liquid and solid propellant rockets.

Heat Transfer Failures

There appear to be at least two basic types of heat transfer failures in combustion chambers. In one type the wall temperature on the gas side T_{wg} exceeds the value at which the material is readily melted or oxidized. The local loss of material and local heating weakens the wall so that the remaining material is inadequate to take the imposed load. This failure is characterized by melting, erosion, or severe oxidation on the gas side surface of the wall. This type of failure occurs primarily in uncooled chambers and uncooled nozzles, but also in cooled chambers or nozzles with relatively thick walls.

The other type of failure is essentially caused by the local inability of the liquid film of the cooling jacket to transfer the heat from the wall to the liquid coolant. This type occurs only in liquid propellant cooled chambers and is characterized by a sudden increase in the local heat flux and wall temperatures, usually resulting in melting of the metal. This breakdown of the liquid film occurs if the wall temperature on the coolant side exceeds the boiling point of the coolant liquid by a considerable margin. A discussion of some of the

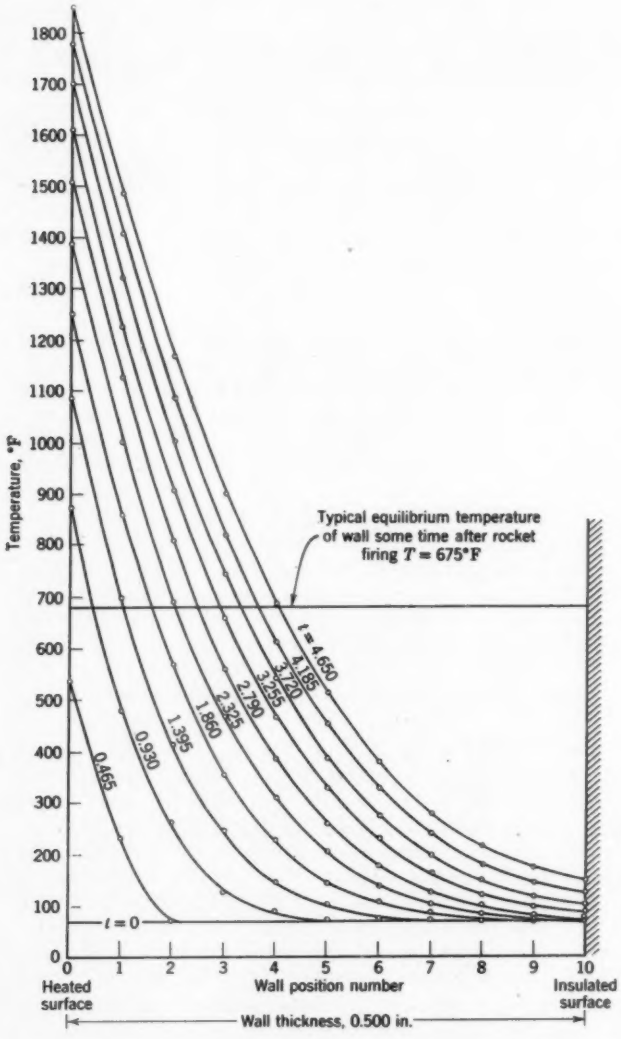

FIG. 3. Typical time-temperature-distance plot for uncooled rocket chamber wall.

factors governing the behaviour of liquid film in forced convection cooling jackets is given in another section of this paper.

Uncooled Combustion Devices

Uncooled walls act essentially as heat sponges and absorb heat from the hot gases. With the aid of experimental data to determine some typical coefficients, it is possible in some cases to predict the transient heating of uncooled walls.

Heat is transferred from the hot gases to the wall, and during operation a changing temperature gradient exists across the wall. The heat transferred from the hot wall to the surrounding atmosphere and by conduction of metal parts to the structure is usually negligibly small. During the combustion process the gas side of the wall is always hotter than the outside wall surface. Each local point within the wall has its temperature raised as the burning process is extended in time. At the completion of the rocket's operation, the wall temperatures tend to equalize. A typical temperature-time-location history is given in Fig. 3. Here the line at $T = 70°$ F. denotes the initial equilibrium condition of the wall before the rocket operates; the various curves show the temperature profile across the wall at successive time intervals after initiation of combustion. The line at $T = 675°$ F. shows an equilibrium temperature of the wall a finite time after cut-off.

In all flying articles weight is at a premium and only sufficient metal is built into the walls to absorb the rejected heat without risking a failure from a weakening of the metal with increasing wall temperature. Thus the rocket designer has to satisfy himself on having the ability to carry all the applied load in the wall at the elevated temperature condition and thus to minimize the weight of the rocket. This condition implies that the heat transferred across the hot surface of the wall (and distributed within the wall by conduction) must be less than the heat-absorbing capacity of the wall material below the critical temperature. If heat transfer to the outside atmosphere is neglected, then this can be expressed in very simplified form:

$$Q\,\Delta t = -\,kA\,(dT/dL)\,\Delta t = W\bar{c}\,\Delta T \quad .. \quad .. \quad (9)$$

Q is the heat in B.Th.U./sec. transferred across area A expressed in square inches. The heat conductivity k depends on the material and the temperature and is expressed B.Th.U./in.²/sec./degree Fahrenheit per linear inch; ΔT denotes the average wall temperature increment in degrees Fahrenheit, dT/dL the temperature gradient of the heat flow near the hot wall surface in degrees per inch, W the weight of the wall in lbs., $\bar{c}$ the average specific heat of the wall material in B.Th.U./lb./degree; and Δt denotes the time increment in seconds. Detailed analysis of this problem requires iterative mathematical treatment as shown below. The simple relation of equation 9, indicates the following trends:

(1) A short operating duration (Δt) should favour uncooled chamber design, because it minimizes the amount of heat absorbed by the wall.

(2) A low value of the conductivity at the hot surface of the wall is desirable. This can be achieved effectively by an insulating layer on the hot gas side. Ceramic linings, special paints, and fuels which deposit a protective coating (e.g., carbon) on the wall are satisfactory for this purpose.

TABLE II. PHYSICAL PROPERTIES OF TYPICAL REFRACTORY MATERIALS FOR ROCKET APPLICATIONS

Material	Typical Composition	Melting Point, °F.	True Specific Gravity	Apparent Specific Gravity	Linear Coefficient of Thermal Expansion, $(°F.)^{-1}$	Thermal Conductivity, B.Th.U./hr. ft.^{2} °F./in.	Specific Heat, B.Th.U./lb. °F.	Resistance to Thermal Shock
Alumina	99·2% Al_2O_3 0·6% SiO_2	3720	3·95	2·9–3·2	0·0000040 at 60° to 1850° F.	17 at 2000° F.	0·28–0·30 at ambient to 3100° F.	Good
Magnesia	97% MgO 1·2% SiO_2	4900	3·58	3·0	0·0000085 at 60° to 2700° F.	18 at 2000° F.	0·29 at 60° to 3270° F.	Good
Zirconia	94·5% ZrO_2	4600–4700	5·6	4·4	0·0000045 at 30° to 2000° F.	6·0 at 2000° F.	0·175 at 60° to 2550° F.	Excellent
Silicon nitride	Si_3N_4	Decomposes in inert atmosphere at 3450	3·44	2·0–2·4	0·00000145 at 1650° F.	—	—	Excellent
Carbon	Essentially C	Sublimates 6402	1·8–2·1	1·03–1·77	0·0000016 to 0·0000029 at 60° to 220° F.	12 to 104 at 2000° F.	0·29 at 60° to 1850° F.	Excellent
Graphite	Essentially C	3681	2·25	1·03–1·91	0·0000010 to 0·0000024 at 60° to 220° F.	240 to 1132 at 2000° F.	0·29 at 60° to 1850° F.	Excellent
Silicon carbide	SiC with complex nitride bond	SiC decomposes at 4010	—	2·77	0·0000037 at 60° to 2750° F.	113	—	Excellent

Typical values are given in Table II. Ceramic nozzle inserts have been used successfully in solid propellant rocket practice.

(3) A high value of the specific heat of the wall material promotes a high heat-absorbing capacity.

(4) A high value of the conductivity of the wall material (other than at the gas surface) will tend to minimize temperature gradients within the wall and thus not only reduce thermal stresses (which are superimposed on the stresses induced in the wall by pressure loads) but also tend to increase the heat-absorbing capacity of the wall material.

(5) A high value of the maximum permissible stress at the highest possible elevated temperatures will permit a large increase in wall temperature without undue weakening of the wall. Unfortunately all metals lose strength as their temperature is increased.

A graphical or tabular method for solving complex mathematical transient heat transfer problems is credited to E. Schmidt who devised a step by step relation procedure, which can be found in some of the references. The values given in Fig. 3 are based on this method.

Steady State Transfer to Liquids in Cooling Jacket

The term regenerative cooling is used for rockets where one of the propellants is circulated through cooling passages around the combustion device prior to the injection and burning of this propellant in the chamber. This is really forced convection heat transfer. The term "regenerative" is perhaps not altogether appropriate here, and it bears little relation to the meaning given to it in steam turbine practice. It is intended to convey the fact that the heat absorbed by the coolant propellant is not wasted but augments the initial temperature and raises the energy level of the propellant as it passes through the injector. This increase in the internal energy of the liquid propellant can be calculated as a correction to the heat of reaction; however, the overall effect on the rocket performance is usually very slight. With some propellants the specific impulse can be 1 per cent. larger if the propellants are preheated through a temperature differential of 200 to 300° F. In small combustion chambers, where the wall surface to chamber volume ratio is relatively large, the temperature rise in the regenerative coolant will be high and the resulting increase in specific impulse can sometimes be more than 1 per cent.

The behaviour of the *liquid film* is the critical phenomenon controlling the wall temperatures in forced convection cooling of rocket combustion devices at high heat fluxes. At least four different types of film appear to exist, as can be interpreted from Fig. 4.

Here the heat transfer rate per unit of wall surface is shown as a function of the difference between the wall temperature on the liquid side T_{wl} and the bulk temperature of the liquid T_l.

1. The normal forced convection region at low heat flux appears to have typical liquid laminar boundary layer. It is indicated by region *A-B* in Fig. 4. Here the wall temperature is usually below the boiling point of the liquid at the cooling jacket pressure. Since the present heat transfer analysis is generally

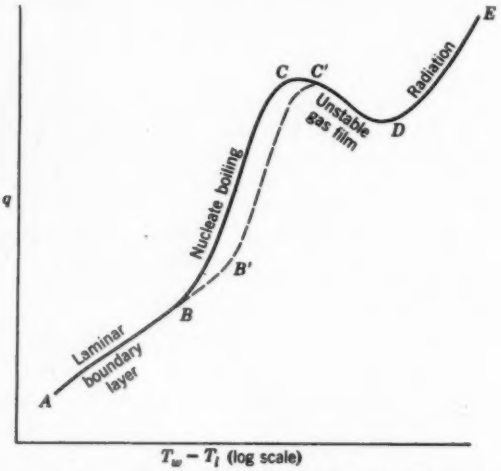

FIG. 4. Regimes of heat transfer in liquid film of coolant fluid.

valid for flow in pipes, conduits, and furnaces, it is adequate for determining the liquid film coefficient in this regime by the usual equation:

$$h_l = 0{\cdot}023\,\bar{c}\,(\dot{w}/A)\left(\frac{Dv\rho}{\mu g}\right)^{-0.2}\left(\frac{\mu g\bar{c}}{k}\right)^{-2/3} \quad .. \quad .. \quad (12)$$

where $\dot{w}$ is the fluid mass flow rate in lbs./sec., $\bar{c}$ is the average specific heat of the fluid in B.Th.U./°*R*-lb., A is the cross-sectional flow area in square inches, D is the equivalent diameter of the coolant passage cross section in feet, v is the fluid velocity in ft./sec., ρ is the density in lb./cu. ft., μ is the absolute viscosity in lb./sec./ft.^{2}, and k is the conductivity in B.Th.U./sec. ft.^{2} °*R*/in. Many liquid-cooled rocket devices operate in this regime of heat transfer.

The helical or annular passages in liquid propellant rocket chambers are often of complex cross section. The equivalent diameter, needed for fluid film heat transfer calculations, is usually defined as four times the hydraulic radius of the coolant passage, divided by the wetted perimenter.

2. When the wall temperature T_{wl} exceeds the boiling point of the liquid by perhaps 10 or 40° F., then small vapour bubbles will form at the wall surface. These small nuclei-like bubbles will cause local turbulence, break away from the wall, and collapse in the cooler liquid. This phenomenon is known as nucleate film boiling. The turbulence induced by the bubbles changes the character of the liquid film and augmented by the vaporization of some of the propellant, the heat transfer rate is increased without a proportional increase in the temperature drop across the film, as can be seen by the steep slope of the curve in Fig. 4. The nucleate boiling region is shown as B-C in this figure. If the pressure of the fluid is raised, then the boiling point is also raised and the nucleate boiling region shifts to the right to B′-*C*′. This film boiling permits

a substantial increase in the heat transfer and often occurs locally in the nozzle throat area, where the heat flux is high.

3. As the heat transfer is increased further, the rate of bubble formation and perhaps also the bubble size become so great that the bubbles are unable to escape from the wall rapidly enough. This region (shown as *C-D* on Fig. 5) is characterized by an unstable gas film and is difficult to obtain reproducably in tests. When a film consisting largely or completely of gas, forms along the hot wall surface, then this film acts as an insulation layer causing a decrease in heat flux and usually a rapid increase in wall temperature, often resulting in a burnout or melting of the wall material. This limiting burnout heat transfer condition is a critical parameter in the design of a rocket; its location on the curve of Fig. 4 appears to be a function of the cooling fluid properties, the presence of dissolved gases, the pressure, and the flow velocity. Several semi-empirical methods have been proposed for predicting the critical burnout conditions. It is essentially caused by a breakdown of the liquid film.

4. As the temperature difference across the film is further increased, the wall temperatures reach values in which heat transfer by radiation becomes important. Region *D-E* is characterized by a very hot wall, an essentially continuous gas or vapour film, and the heat is transferred almost completely by radiation. This region is of little interest to the rocket engineer.

The general phenomenon of changing film behaviour is not fully understood; however, its general pattern is fairly well defined.

In addition to the liquid film phenomenon described above, chemical changes in the liquid can seriously influence the heat transfer from hot walls to liquids. Cracking of the fuel, with an attendant formation of gas or the formation of solid material which is deposited on the hot wall surface, tends to reduce the maximum heat flux and thus promote failure more readily.

The heat-absorbing capacity of the coolant has to be such that boiling is permitted locally but the bulk of the coolant does not reach this boiling condition. The heat rejected by the surface A of the hot walls, as given by equations 2 to 6, has to be less than that permitted by the temperature rise in the coolant

$$qA = Q = \dot{w}c\,(T_1 - T_2) \quad .. \quad .. \quad .. \quad (13)$$

where $\dot{w}$ is the coolant flow rate, c the specific heat of the liquid, T_1 is the initial temperature of the coolant as it enters the cooling jacket, and T_2 is the final temperature of the coolant in the jacket. Q is the rate of heat absorption per unit time; q is this same rate per unit heat transfer area A. T_2 must be below the boiling point prevailing at the cooling jacket pressure. Typical values of the heat absorption by a coolant are shown in Fig. 2.

Radiation

The heat radiation transmitted by the mechanism of radiation depends primarily on the temperature of the body. The second law of thermodynamics can be used to prove that the radiant energy E is a function of the fourth power

$$E = \epsilon\sigma AT^4 \quad .. \quad .. \quad .. \quad .. \quad (14)$$

The energy E radiated by a body is defined as a function of the missivity ϵ, which is a dimensionless factor for surface condition, material properties, or geometry relations, the Stefan-Bolzmann constant σ (0.1713 B.Th.U./ft.^{2} hr. $°R^4$), the surface area A, and the absolute temperature T. At low temperatures (below 1,000° F.), radiation accounts for only a negligible portion of the total heat transfer in a rocket device and can usually be neglected. In rocket combustion devices, gas temperatures are between 3,500 and 7,000° R., and radiation is believed to contribute between 5 to 25 per cent. of the heat transfer to the chamber walls.

The absorption of radiation follows essentially the same laws as those of emission. A highly reflective surface on the inside wall of a combustor will tend to reduce absorption and to minimize the temperature increase of the walls.

The radiation in combustion devices comes usually from the hot reaction products. These gases are not as effective a radiator as solid matter; however, at the high combustion temperatures the contribution of gas-emitted radiation to the overall heat transfer is significant.

Gases with symmetrical molecules, such as hydrogen, oxygen, nitrogen, etc., have been found not to show emission bands in those wavelength regions of importance in radiant heat transfer. Also they do not absorb radiation and do not contribute energy. Heteropolar gases such as water vapour, carbon monoxide, carbon dioxide, hydrogen chloride, hydrocarbons, ammonia, oxides of nitrogen, and the alcohols have strong emission bands of known wavelengths. The radiation of energy of these molecules is associated with their quantum changes in their energy levels of rotation and interatomic vibration. In general the radiation intensity of all gases increases with their volume, partial pressure, and the fourth power of their absolute temperature. For small thrust chambers and low chamber pressures, radiation contributes only a small amount of energy to the overall heat transfer.

When the combustion products contain clouds of finely divided solid matter, then their radiation power is generally increased by a large factor (two to ten times). There are two types of luminosity, depending on the size of particles of solid matter suspended in the gases: those flames containing relatively large particles are essentially opaque, and the gases with soot particles that are very small, have flames that are essentially semi-transparent. The radiation behaviour of these two types is different. In solid propellant rockets it is possible to obtain gases with entrained solid particles of fair size (above 0·1 micron). The incomplete burning of many hydrocarbons, such as aromatic hydrocarbon components, produces radiant soot in the flame with particles of very small size.

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(3) C. M. Beighley and L. E. Dean, "Study of Heat Transfer to JP-4 Jet Fuel," *Jet Propulsion*, **24**, (3), May-June, 1954.
(4) S. Greenfield, "Determination of Rocket Motor Heat Transfer Coefficients by the Transient Method," *Journal of the Aeronautical Sciences*, **18**, (8), August, 1951.
(5) H. Ziebland, "Heat Transfer Problems in Rocket Motors," *J.B.I.S.*, **14**, (5), September-October, 1955.

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(10) S. S. Penner and S. Weinbaum, "Some Considerations of the Effect of Radiation on the Performance of Liquid Fuel Rockets," *Journal of the Optical Society of America*, **38**, (7), July, 1948.
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DATA SHEETS

By K. W. Gatland, F.R.A.S.

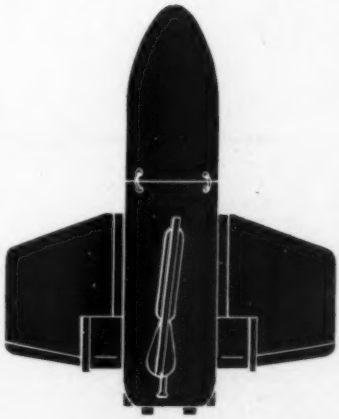
SS-10 (SNCAN 5203) (France)—S.N.C.A. du Nord, Châtillon.

(*a*) Role: Surface-to-surface (anti-tank).
(*b*) Origin: 1950.
(*c*) Length: 2 ft. 9·5 in.
(*d*) Diameter, body (max.): 7·5 in.
(*e*) Diameter, over fins: Not applicable.
(*f*) Diameter, over wings: 2 ft. 5·25 in.
(*g*) Weight, launching: 33 lb.
(*h*) Weight, propellant: not available.
(*j*) Weight, payload: not available.
(*k*) Thrust: not available.
(*l*) Velocity (max.): sub-sonic.
(*m*) Altitude: not applicable.
(*n*) Range: 4,920 ft.
(*p*) Propellant: solid propellant.
(*q*) Feed system: not applicable.
(*r*) Control: Command signals transmitted through fine wires, unwinding from bobbins on wings.
(*s*) Booster: not applicable.

Notes: Spins slowly in flight, stabilised by high-speed gyro and controlled from optical sight by electrical impulses transmitted by wires to spoilers on wings.

Propelled by two solid rockets fired successively: first unit launches and accelerates missile to designed speed; second unit acts as sustainer motor. Missile is fitted with a tracer, guidance being effected by visual alignment on target by means of a joystick, controlling direction and altitude, after cruising motor has started to function.

Warhead consists of a shaped charge (penetration approx. 15·6 in. armour plate). Instantaneous inertia fuse fitted with safety arming device operated by pressure of gas from cruising motor after launching. Warhead attached to missile by means of three toggle fasteners.



SS-10

The type 5202 has an inert head and is used for training purposes.

Container forms launcher when unfolded.

Tested in 1953 by U.S. Army Ordnance.

Development model employed on Potez 75, light attack monoplane (air-to-ground).

REFERENCE

(1) "France's Guided Missiles," *Inter Avia*, January, 1956, pp. 50-51.

RHEINBOTE (Germany)—Rheinmetall-Borsig.

(*a*) Role: Surface-to-surface, long-range bombardment.
(*b*) Origin: 1943.
(*c*) Length: 37·4 ft. (overall), 11·4 ft. (step 1), 11·4 ft. (step 2); 13·1 ft. (step 3).
(*d*) Diameter, body (max.): 9·75 in. (step 1), 9·75 in. (step 2), 6·30 in. (step 3).
(*e*) Diameter, over Fins: 2 ft. (step 1), 2 ft. (step 2), 1 ft. 5 in. (step 3).
(*f*) Diameter, over wings: not applicable.
(*g*) Weight, launching: 3,781 lb. (total), 937 lb. (step 1), 871 lb. (step 2), 441 lb. (step 3).
(*h*) Weight, propellant: 309 lb. (step 1), 309 lb. (step 2), 132 lb. (step 3).
(*j*) Weight, payload: 88 lb. (warhead, 3rd step).
(*k*) Thrust: 12,348 lb. × 5 sec. (step 1), 12,348 lb. × 5 sec. (step 2), 7,563 lb. × 3·5 sec. (step 3).
(*l*) Velocity (max.): 3,660 m.p.h. (final step).
(*m*) Altitude:
(*n*) Range: 136 miles (65 deg. elevation).
(*p*) Propellant: solid, diglycol-dinitrate.
(*q*) Feed system: not applicable.
(*r*) Control: dynamically stable by virtue of tail fins on each step and high initial acceleration.
(*s*) Booster: 6-finned: 6·2 ft. long × 18 in. dia.; thrust: 83,700 lb. × 1 sec.; weight: 1,432 lb. (540 lb. propellant).

Rheinbote

NOTES: First large solid-propellant step-rocket. Launched from mobile ramp.

Used in a limited number by the Germans in the closing months of the Second World War from Zwolle (Holland) against Antwerp (Belgium).

REFERENCE

(1) *Development of the Guided-Missile*, by K. W. Gatland (2nd edition), Iliffe & Sons, London, 1954, pp. 148, 149, 268, 269.

Silhouettes by M. F. Allward.

GASSIOT VEHICLE (Great Britain)—Royal Aeronautical Establishment, Farnborough.

(*a*) Role: Research, upper atmosphere.
(*b*) Origin: 1956.
(*c*) Length: 25 ft. 7 in.
(*d*) Diameter, body (max.): 17·4 in.
(*e*) Diameter, over fins: 7 ft. 3 in.
(*f*) Diameter over wings: not applicable.
(*g*) Weight, launching: 2,500 lb. approx.
(*h*) Weight, propellant: not available.
(*j*) Weight, payload: 150 lb.
(*k*) Thrust: 11,500 lb. × 30 sec.
(*l*) Velocity (max): 3,900 m.p.h.
(*m*) Altitude: 104 miles.
(*n*) Range: not applicable.
(*p*) Propellant: plastic, slow-burning (Raven unit).
(*q*) Feed system: not applicable.
(*r*) Control: none, dynamically stable by virtue of three fixed tail-fins.
(*s*) Booster: not applicable.

Gassiot Vehicle

NOTES: Launched from lattice tower 80 ft. high, pivoted near its centre from triangular mounting to allow limited degree of elevation and traverse.

Experiments devised and flight instruments built by a number of University departments and by R.A.E., under general guidance of Gassiot Committee. Research includes:—

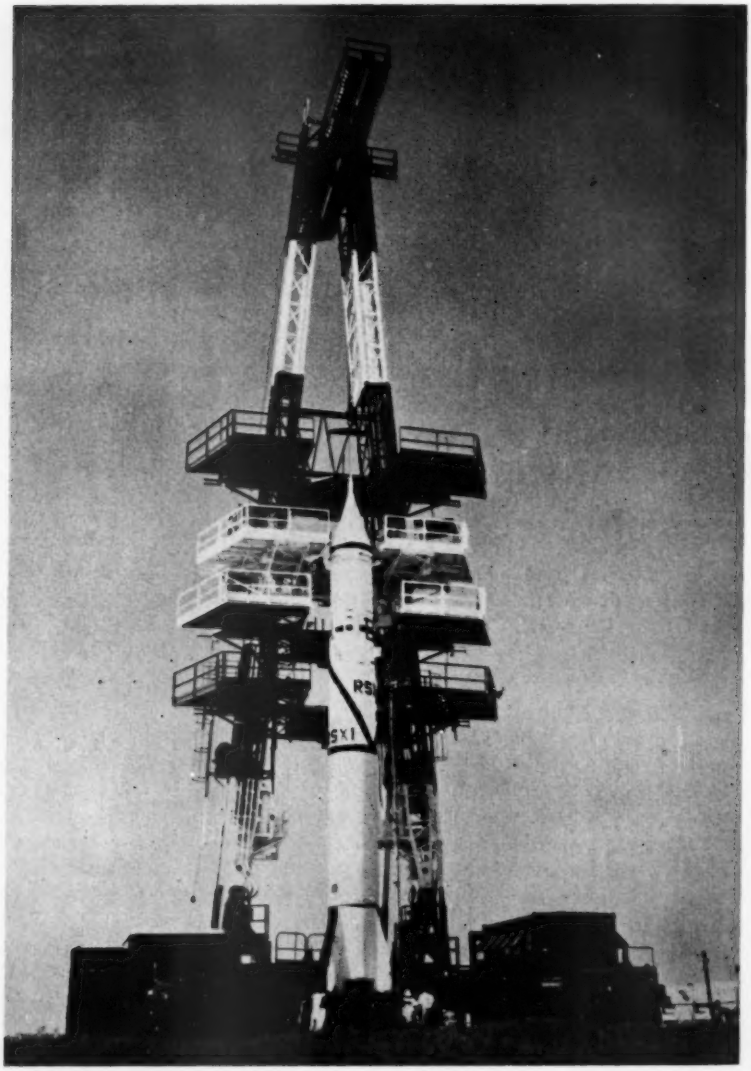
(1) Wind, temperature and pressure by grenade method; observations made at a number of points on the ground of series of grenades fired from rocket at approximately equal intervals along trajectory (University College, London).

(2) Mass Spectrometer for ion sampling (University College, London).

(3) Study of ionspheric conductivity and relative concentration of electrons to negative ions by sampling ionization chamber fitted with a pulse method of distinguishing between ions and electrons (University of Birmingham).

(4) Photo-electric photometers and filters to study relative intensities of certain spectral lines of molecular oxygen and the altitude of night air-glow in regions of oxygen green line and the Meinel OH bands; also study of emission resulting from release of sodium and potassium vapour at high altitudes. (Queen's University of Belfast.)

(5) Two methods—continuous-wave Doppler and pulse method—of measuring electron densities in ionsphere and in correlating these with measurements made from the ground (University College, Swansea).



Army's Redstone missile in service station at Patrick Air Force Base, Florida.

(6) Measurements of skin temperature and altitude of rockets; also design of sunseeker for carriage in the rocket, to be used for measurements of solar radiation and for obtaining ultra-violet pictures of sun at high altitudes. (Royal Aircraft Establishment, Farnborough.)

First experimental firing at Woomera, probably in September or October, 1956. Several rockets to be launched in Australia and Great Britain during International Geophysical Year (July, 1957–December, 1958). A second version, under development 1956, is expected to reach 130 miles.

REFERENCE

(1) "Rocket Exploration of the Upper Atmosphere," by F. E. Jones and H. S. W. Massey, *Nature*, April 7, 1956, pp. 643-647.

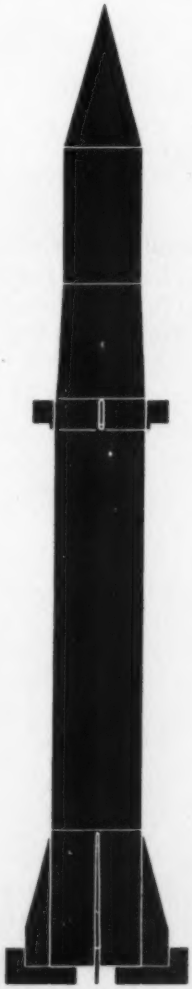
JUPITER A, Formerly **REDSTONE XSSM-A**

(United States of America)—Chrysler Corporation, Detroit, Michigan.

(*a*) Role: Surface-to-surface, long-range bombardment.
(*b*) Origin: 1952.
(*c*) Length: 60 ft. approx.
(*d*) Diameter, body (max.): 60 in.
(*e*) Diameter, over fins: 9 ft. 6 in.
(*f*) Diameter, over wings: not applicable.
(*g*) Weight, launching: not available.
(*h*) Weight, propellant: not available.
(*j*) Weight, payload: not available.
(*k*) Thrust: 70,000 lb. (approx.).
(*l*) Velocity (max.): 4,000 m.p.h. (approx.).
(*m*) Altitude: 80 miles (for max. slant range).
(*n*) Range: 230 miles.
(*p*) Propellant: not available.
(*q*) Feed system: hydrogen-peroxide turbo-pump.
(*r*) Control: similar to German A-4.
(*s*) Booster: not applicable.

Provisional silhouette.

NOTES: Developed at Redstone Arsenal, Huntsville, Alabama, by U.S. Army Ordnance under technical supervision of Dr. Wernher von Braun (co-designer of German A-4 rocket). Completed test firings at White Sands in 1952.



Redstone

Rocket engine supplied by North American Aviation. Conventional H.E. or nuclear warhead. Fin and rudder surfaces reminiscent of Wasserfall. Warhead separates from missile body at end of propulsion phase.

REFERENCE

"Current progress in guided weapons," *The Aeroplane*, May 20, 1955, pp. 665 and 667.

TECHNICAL REVIEW

Compiled by E. T. B. SMITH, B.Sc., A.F.R.Ae.S., G.I.Mech.E.

With this issue it is intended to indicate the source from which the compiler has obtained his information. To begin with, the following abbreviations will be used—AW—Aviation week; JP—Jet Propulsion; USIS—United States Information Service. Additions will be made as needed.

Satellite Vehicles

(*a*) *Initial Test Programme*

Details are now avaialable (USIS, January 27, 1956) of the programme of experiments for the U.S. National Committee for the IGY, to be carried out in the Vanguard firings. They are:—

(i) Density of the outer atmosphere. Observations of the satellite will permit the determination of air density at high altitudes with an accuracy impossible otherwise.

(ii) Composition of the crust. Careful observation of variations in the orbit may provide information on the composition of the Earth's crust, for the non-uniform distribution of mass in the crust may affect the satellite's orbit.

(iii) Geodetic determinations. Measurements of the orbit, related to (ii), should yield information on the oblateness of the Earth, and this may provide a better picture of the Earth's shape. Synchronized observations may permit improvements in determination of longitude and latitude.

(iv) Temperature. Temperatures within the satellite and on its surface will be measured.

(v) Pressure. Pressures within the satellite will be measured.

(vi) Meteoric Impacts. A detection system for ascertaining meteorite impact on and penetration of the satellite shell is included in current plans.

(vii) Ultra-Violet Radiation. Instrumentation to measure the extreme ultra-violet radiation from the Sun is included in present plans. Lyman-α measurements will yield data on solar radiation of this type and may provide information on the presence of hydrogen in interstellar space.

(viii) Cosmic rays. Measurements of cosmic ray intensities will be made, including the low energy particles masked from the Earth by its atmosphere.

Of these experiments, the first seven, or the first six and the eighth, can be achieved in the first satellite, and other experiments are being considered for additional satellites.

(*b*) *Launching Trajectory and Orbit*

The orbit of the satellite has been tentatively established: an inclination of 40° from the Equator has been chosen, so that the orbit will shift within apparent latitudes of 40° on either side of the Equator. The launch will be made from Patrick Air Force Base on the east coast of Florida, at Cape Canaveral (long. 28° 28′ N., lat. 80° 23′ W.)

The launching flight path is expected to consist of a vertical ascent for about the first mile after launching, followed by a gradual inclination towards the horizontal. The first stage rocket will burn out and drop off at 30 to 40 miles altitude, when a velocity of 3,000 to 4,000 m.p.h. is reached, and the second stage will burn out at 130 miles altitude and about 11,000 m.p.h. and coast under its guidance (sic) into the satellite trajectory. At about 300 miles altitude, the third motor will fire and accelerate the satellite vehicle to its orbital velocity of about 18,000 m.p.h.

(*c*) *Technical Details of Vehicle Stages*

We gather that, during launch, aerodynamic heating will raise the temperature of the nose cone to about 540° C. The first stage motor, made by General Electric, will produce a thrust of 27,000 lb. wt. for about 140 sec., using as propellents liquid oxygen and a mixture of ethyl alcohol, gasoline and silicone oil. Propellent supply will be provided by turbopumps driven by decomposed H.T.P. The second stage will have an engine, built by Aerojet, which will run on fuming nitric acid and asymmetrical dimethyl hydrazine, supplied from helium-pressurized tanks. The third stage solid propellent rocket motor is now known (USIS April 6, 1956) to be in the hands of the Grand Central Rocket Co., California, and Allegheny Ballistics Laboratory of Maryland.

Other contractors now known (USIS, April 6, 1956) are Minneapolis—Honeywell Regulator Co. of Minnesota, for the gyro reference system, and Vickers Inc. of Minnesota for the autopilot, working from the gyro system and controlling the complete vehicle.

In respect of gyro control systems, reference should be made to a series of articles (AW, January 2, 9, 16 and 23, 1956) on the properties of inertial guidance systems and some methods of designing them. While the articles are too long to summarize in detail, they are most instructive in a field of vehicle design which is normally hedged around with security, and have a lot of information on the applications of inertial guidance, which is independent of radio control or celestial guidance. Basically, such systems are composed of gyros, accelerometers and integrators, but great difficulties are experienced in getting sufficient accuracy and freedom from drift to make them worthwhile.

Present plans indicate (AW, January 30, 1956) that the first instrumented satellite will weigh 21·5 lb., with a diameter of some 20 inches. It will be protected against aerodynamic heating by a disposable nose cone when mounted

on the launching vehicle. To stabilize the flight of the third stage rocket into the final orbit, axial spin will be imparted before ignition, so that the satellite will start in its orbit with this spin: however, eddy currents induced by the Earth's magnetic field in the satellite's metallic skin will soon reduce this to zero.

Due to its being alternately in sunlight and shadow, temperatures within the satellite will vary between —20° C. and 200° C.

(*d*) *Applications of Satellite Vehicles*

R. P. Havilland, of the (U.S.) General Electric Co., has discussed the general utility of a satellite vehicle (JP, January, 1956). He gave the fields of potential usefulness as: Mapping and Geodesy, Communications, Weather charting and forecasting, Research, and the development of Space flight. He also listed the fields of vision of the Earth from orbiting vehicles at different altitudes: at 57 miles, 8 per cent. of the Earth is visible, at 257 miles, 17 per cent. is visible (an amount equal to the whole U.S.A.), while at 22,300 miles, 50 per cent. is visible, during each revolution of the satellite. However, these values will be somewhat reduced (to about three quarters) by atmospheric absorption and refraction near the edges of the field of vision. The speaker covered Mapping and Geodesy by the mention of triangulation surveys, which could be used to tie together the normal surveys on the continents, and by an indication that a determination of the absolute value of "g" might be made. Photographic mapping methods would be used, and it was stated that an 8 in. focal length camera, using 5 in. wide film, could completely cover the Earth in daylight with 900 photographs, with the satellite in a polar orbit. If physical recovery of the camera could not be achieved, then television or facsimile transmission would have to be used: television with 400 lines per frame would reduce the effective resolving power of the 8 in. lens by a factor of one fifth, while facsimile (1,000 lines per inch) would not cause any reduction in the resolution.

In the Communications field, two uses could be made of the satellite. First, it could be used as a relay station for v.h.f. signals which normally can be used only on "line-of-sight" ranges on the Earth. Extensive use of microwave channels would ease the telephone and telegraph business considerably. Secondly, the satellite could relay television and f.m. broadcasts to regions not normally coverable by ground systems at an economic expense.

For Weather charting and forecasting, the satellite could have scanning gear which would view the earth's atmosphere and show the location of clouds and frontal systems, precipitation and storm areas. This might permit the discontinuance of the various weather ship, iceberg patrol and hurricane location services.

Sunfollowers as Orientation Devices

A paper by Terwilliger and Granros describes (JP, January, 1956) a device, built for the U.S. Naval Research Laboratory, which tracks the Sun so that sunlight is constantly focussed on the entrance pupil of a spectrograph located in the nose of a sounding rocket, even though the rocket may be pitching and rolling after burn out. The instrument described consists of two pairs of photocells which, in conjunction with amplifiers and servo motors, enable the

Sun to be sought, and, when locked on, to be followed within $\pm$ 0·25°. The spectrograph is rotated by the servos in elevation and azimuth. The follower has been flown in two Aerobees and a Viking, and, in the case of the latter, enabled useful spectrographic data to be obtained for 80 seconds of flight above 30 miles, even though the vehicle rolled at nearly 180 r.p.m. at 120 seconds after take off.

It may be pointed out that the measurement and control of the altitude of a satellite is not easy, but is essential if instruments are going to do what they are supposed to do: for this reason, it may well be necessary to include sun-followers or gyro systems in satellites at the expense of instrument payload.

Proposal for a Manned Space Station

A satellite vehicle as large and complicated as the well known proposal of Dr. von Braun has been suggested by D. C. Romick of Goodyear Aircraft Corp. (JP, February, 1956). The vehicle is to be established in its orbit by ferry rockets which carry passengers and materials to the steadily growing station. The first two ferries are to be used to form as the basis of the station structure. Romick estimates that about three years would be needed to build a vehicle with a volume of 3,000,000,000 ft.^{3}: its shape would be that of a cylinder with a large wheel rotating at one end, to give some artificial gravity to those people who feel they need it. The cylindrical portion is to be 1,000 ft. in diameter and 3,000 ft. long, and the wheel, rotating at 1 r.p.m., is to be 1,500 ft. in diameter and 40 ft. thick. The station will be used for the study of solar energy, earth reconnaissance, the guidance of outgoing space ships, and for recreational purposes. No details of construction or cost are available.

A Russian Satellite Vehicle Proposal

Professor K. Stanyakovich of the U.S.S.R. Academy of Sciences Commission on Interplanetary Communications has reported (JP, March, 1956) that Russia will be prepared to launch a larger and more advanced satellite than the type proposed by the U.S.A., shortly after the U.S. satellite is launched. He also stated that directed explosions could be used to advantage in establishing satellites, but unofficial reports indicate that Soviet work is along more conventional lines. The present Russian programme is reported to have reached the development of a final stage after completion of initial development of the first two stages. Gross weight will be between 75 and 100 tons, with a first stage thrust of 120 tons.

High Altitude Balloons and Aircraft

Mention has been made previously in Technical Review of high-altitude balloon flights for cosmic ray investigations, and a description is now available (AW, October 10, 1955) of the work done by General Mills, Inc., in Minneapolis, Minnesota. This firm is using polyethylene balloons with volumes of between two and three million cubic feet, which ascend to over 100,000 ft. for five or six days: these balloons form the Skyhook programme, which has been largely responsible for the "flying saucer" reports of recent years.

Considerable research was undertaken by General Mills Inc. on the design of the envelope shape and the choice of material for it, and it appears that the appropriate shape is that of an inverted pear, almost flat on top and sloping conically to a point at the bottom. The material chosen was polyethylene, heat sealed at the seams, in thicknesses between 0·0005 in. and 0·0025 in. The largest balloon made in this manner is the Super Skyhook, with a length of 282 ft. and a capacity of 3·2 million cubic feet of helium, which uses about 3 acres of plastic sheet and weighs 850 lb. It has reached 117,000 ft.

Difficulties associated with the operation include those of handling during launching, for the sail area of the balloon is very large and even light winds make the envelopes whip and sway and become punctured. Fifteen miles per hour represents the maximum wind speed at which launchings can normally be made, and 22 m.p.h. is the top limit. In flight, a control must be kept on the rate of ascent, which must not exceed 800 ft/min., to avoid the balloon being shattered when passing through the tropopause where temperatures of —60° C. may be encountered. At such temperatures the polyethylene skin is very brittle, and, if the balloon flutters during ascent, or if there is excessive air turbulence or wind shear, failure may readily occur.

In order to maintain height even though the balloons are in the dark overnight, when they lose heat and collapse slightly and so tend to sink, controlled ballasting is needed. First attempts, using sand, were unsatisfactory, since the residual moisture froze and bound the sand together so that the sand could not be discharged. Kerosene was then tried, but it was superseded by magnetically controlled steel shot, in which a permanent magnet retains the shot during normal flight. An aneroid device actuates an electromagnet at excessive heights, which neutralizes the field of the permanent magnet and allows shot to be released.

The balloon flights by Winzen Research Inc. used balloons of almost identical size and construction.

Balloon-sondes are being launched from the U.S. Navy's air facility at Oppawa, Japan, to collect weather information in their flights across the Pacific, and one such ballon is reported to have landed in Mississippi, U.S.A. The balloons are made of polyethylene, are 39 ft. in diameter, and carry weather instruments, a 50 watt radio transmitter, and power supplies. There is enough ballast to ensure flights between 3 and 6 days, during which time the balloon may travel 10,000 miles. The instruments are lowered by parachute as soon as the balloon drops below 30,000 ft.

A reported study by the U.S. Office of Naval Research shows that it would be possible to build, with a motor at present available, a manned rocket aircraft which could attain an altitude of 125 miles at a speed of about 3,700 m.p.h. The aircraft would be strictly a research vehicle, to study such problems as weightlessness. An advertisement (AW, January 30, 1956) indicates that North American Aircraft have been given a contract to build such an aircraft, designated X-15.

New Propulsion Devices

In a paper delivered to the First International Conference on the Physics of Jet Propulsion, in Freudenstadt, S. Germany, Dr. Theo. Peters

of Siemens discussed a so-called plasma-flow rocket. He had experimented with a water-stabilized high-powered electric arc, with the wall of the heating chamber formed by a vortex of water, which would be capable of a very great thermal loading: with a maximum stagnation temperature of 52,000° K., an arc power of 450 kW. and an arc diameter of 2 mm., there is an energy release of 11,000 kW./cm.^{3}, or 10^{13}kcal./m.^{3}hr. The plasma beam leaving the chamber may have a very high speed, but this has not been investigated at such high energy levels.

For high pressure operation, a water stabilized arc was developed in which an energy release of 350 kW./cm.^{3} at a pressure of 100 atm. was obtained for short periods, to ascertain the radiation properties of hydrogen under high pressure. A high pressure chamber has been made in which the arc anode has an aperture in the form of a de Laval nozzle. A boiler, partly filled with water, has a pressure of 50 atm. produced in it, and the mean temperature on the arc axis is 12,500° K. after ignition of the arc. The water vapour plasma,* heated to this temperature in the arc, is expanded through the nozzle and passes out into the atmosphere at a speed of 6,500 m./sec. and a temperature of 8,000° K.

The Rocket Fuels Division of the Phillips Petroleum Co. has produced a rocket motor for boost applications which is stated (AW, March 19, 1956) to have the highest thrust ever produced by a single rocket motor of any type, which means that the thrust must be well above the 125,000 lb. wt. level quoted for individual units of ICBM engines. However, North American Aircraft have been firing something very big (possibly for the Atlas ICBM), since they have been awakening many people in Southern California by their midnight firings, lasting up to 40 seconds.

Test Facilities and Establishments

A description has recently been given (AW, February 13, 1956) of the work in hand at the Jet Propulsion Laboratory of the California Institute of Technology at Pasadena. Aerodynamic work at JPL includes testing of missiles and ramjets, for which two supersonic tunnels are used, one of 12 in. × 12 in. working section with M = 4, and the other of 18 in. × 20 in. with M = 5. A new tunnel (by 1957) will be 21 in. square with M = 8. There is also research proceeding on boundary layer transition, surface roughness effects, and heat transfer.

The Electronics division covers guidance, transistor applications and telemetry, while the Guided Missile Engineering division takes the findings of the research divisions and makes missiles using these data: work is mainly devoted to liquid propellent propulsion development, ground launching gear, structural design, and service liaison. This group has been responsible for WAC-Corporal, Bumper, and Corporal.

Development is done by the Rocket and Materials division on propellents and metallic and refractory materials of construction, and research in combustion and instrumentation is performed by the Applied Research division. JPL

* "Plasma" in this connection is a term given to a gas under considerable electric stress, in which may be found, simultaneously, an "electron" gas, a "positive ion" gas, and a molecular gas.

employs about 1,200 people, 350 being engineers and scientists, and 400 being technicians: the laboratory facilities cover 80 acres, and the installation is valued at $16,000,000.

There is also a new facility for liquid propellent motors being built by Aerojet-General at their solid propellent plant at Sacramento, California (AW, March 19, 1956). There are to be five test stands, which will accommodate motors of up to 1,500,000 lb. wt. thrust, and they are being built on the edge of a 40 ft. ravine on the site, in reinforced concrete with steel superstructures from which the exhaust jets flow vertically downwards. The steel deflector plates, 15 ft. × 20 ft., have large water spray nozzles for cooling purposes.

Control rooms are completely windowless and are banked with earth, and observation of the motors during firing is by closed circuit television. Instrumention includes large quantities of IBM and Remington Rand equipment for data handling.

There are bulk storage facilities for liquid oxygen and other propellants near the stands, but tankage for the actual firings is installed in rooms at the bottom of each stand inside the structure.

A buffer area a mile in depth surrounds the test stands and service buildings, and the administration building is 3 miles away from the stands: there is also a 2½ acre fabrication building for component work and development. Eventually, 5,000 people will be employed at the Sacramento plant: these will include the workers on the solid propellant production lines.

The first guided missile training centre of the Royal Navy has been opened at Portsmouth. At the opening ceremony it was revealed that H.M.S. Girdle Ness, the landing craft maintenance ship to be used for the testing of the Navy's guided weapons, will be commissioned in midsummer, 1956.

Correction

It was stated in a recent Technical Review that an Aerobee-Hi had set an altitude record of 180 miles, but apparently this claim was due to observational enthusiasm (JP, January, 1956). The record for single stage rockets is still held by Viking 11 at 158 miles, at the time of writing.

I.A.F. TECHNICAL PROCEEDINGS.

Copies of Technical Papers presented at past I.A.F. Congresses are still available as follows:—

Stuttgart 1952	*Probleme aus der Astronautischen Grundlagenforschung.* H. H. Kolle, et al. G.f.W., Stuttgart. pp. 256. £2 1s. 8d.
Zurich 1953	*Space Flight Problems.* S.A.A. pp. 224.
Innsbruck 1954	*Bericht uber den V International Astronautischen Kongress.* F. Hecht (Ed.) O.G.f.W., Wein pp. 307. £4 18s. 6d.
Copenhagen 1955	To be published in due course.

The Proceedings of the London 1951 Congress were published in a 74 pp. booklet, "The Artificial Satellite" L. J. Carter (Ed.) 5/–, but this is now out of print.

The above Technical Papers as well as new books on astronautics, astronomy and other sciences, especially those advertised in the Society's Book List, can be obtained from:—

INTERPLANETARY PUBLISHING CO.,
12, Bessborough Gardens, London S.W.1.

NOTES AND NEWS

Reduced Fees Payable by Students

In accordance with the amendment to the Society's Articles (Article 5 (f)) approved at the Annual General meeting held on April 21, 1955, the Council has recently adopted the following new Bye-Laws:—

(*a*) That Bye-law 10 be renumbered 10 (a).

(*b*) That a new Bye-law, numbered 10 (*b*), be inserted as follows:—

Full-time students at any university, college or school recognized by the Council shall pay a reduced fee of £1 1s. per annum in lieu of the membership and fellowship subscriptions detailed in article 5, subject to the following provisos:—

(1) That a certificate from the educational or other competent authority should accompany such application to pay reduced fees each year, indicating that the applicant is a full-time student.

(2) That a total time limit for the acceptance of such certificates be for a maximum of three consecutive years.

(3) This Bye-law shall become effective from January 1, 1957.

Lending Library List

A revised list of books available in the Society's postal Lending Library has now been prepared and copies can be obtained on application.

At the present time, books may be borrowed only by Fellows of the Society resident in the United Kingdom and should not be retained for a period longer than one month unless application for renewal is made.

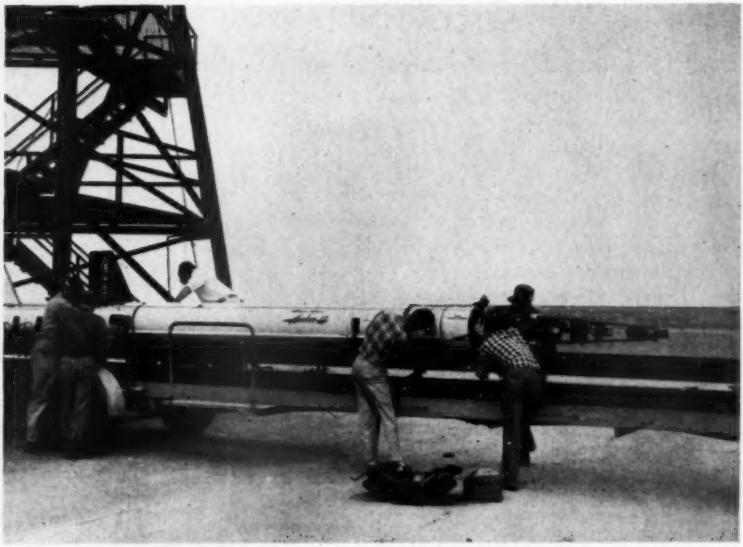
1956/57 Lecture Advertising Posters

The Council is endeavouring to obtain the greatest possible publicity for the 1956/57 Lecture Programme, for not only will this take place during a period of intense activity in the field of astronautics, but the programme itself is probably one of the best we have ever arranged.

A large number of special display posters are being prepared for distribution to Universities, Colleges, Libraries, etc., and the Secretary would be pleased to hear from any member who could arrange for these notices to be displayed also in Industrial firms, Technical Institutes, and elsewhere likely to attract attention, besides arranging for short notes to appear in staff magazines, the local press, etc. Any member who might be able to assist in arranging such publicity is invited to write to the B.I.S. Secretary for further details.

1957/58 Lecture Programme

The Council will shortly consider the 1957/58 Lecture Programme, and would be interested to hear from any members who have suggestions for possible talks. Such suggestions should also include recommendations as to possible speakers, and in the case of Branch members, communications can be sent to the local Branch Secretary who may be able to incorporate some of the suggestions into the Branch Lecture Programme.



An instrumentation crew at Holloman Air Development Centre, New Mexico works on the Aerobee-Hi, a new upper atmosphere research rocket. Intricate instruments, designed to obtain information on the atmosphere at heights up to 135 miles are carried aloft by the Aerobee-Hi rockets. The instruments are contained in a special nose cone which is blown free of the rocket and parachuted to the ground. The new rocket reached an altitude of 123 miles on its first launch recently at Holloman.

Journal Binding

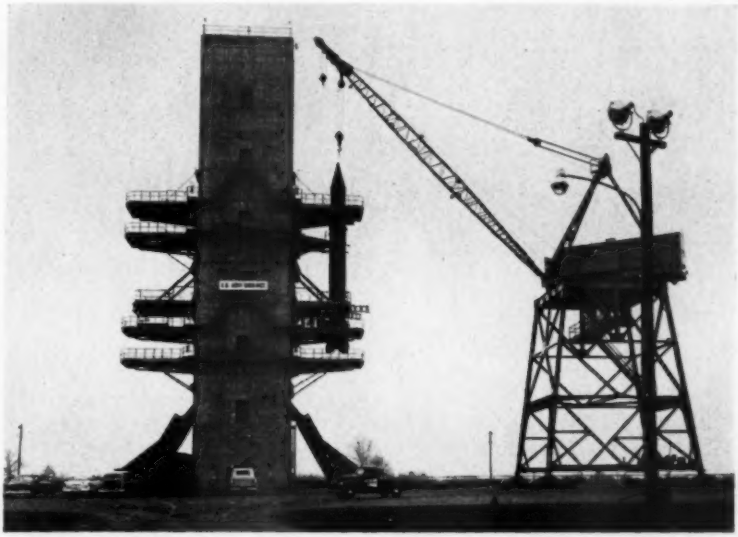
We are advised by our printers that they have been forced to increase the price for binding members' copies of the Journal to 9s. 6d., per volume owing to the general increase in costs.

Members who wish to have their copies bound should send them, together with the index, direct to W. Heffer & Sons, Ltd., Hills Road, Cambridge, with a remittance for this amount. Please do not send them to the Society.

Press Release

The Society has initiated a new system whereby press handouts will be distributed on a wide basis, and the co-operation of members is requested who are in a position to make use of such information e.g., by including short reports in the local press, news, club, and industrial magazines, etc. It is planned to issue releases not only affecting the Society, but also on other matters of interest in the fields of rocket development and spaceflight. A special distribution list is being maintained by the Society for this purpose.

The first release, already issued was concerned with the design studies for an orbital satellite, and the second release, due for issue next month, will be concerned with our new magazine, "SPACEFLIGHT".



Army's Redstone missile in static firing stand at Redstone Arsenal, Huntsville, Alabama (day shot).

The Council has also formed a small Committee to deal with the question of relations with the press, this Committee consisting of Messrs. R. A. Smith, K. W. Gatland, L. J. Carter, and P. Moore. Suggestions and information on the work of this Committee can be obtained on application to the B.I.S. Secretary at 12, Bessborough Gardens, London, S.W.1.

Personnel Changes

It is reported that Mr. J. E. P. Dunning has been appointed as Chief Superintendent, Rocket Propulsion Department, R.A.E., with effect from December 1, 1955. He succeeds Dr. T. P. Hughes.

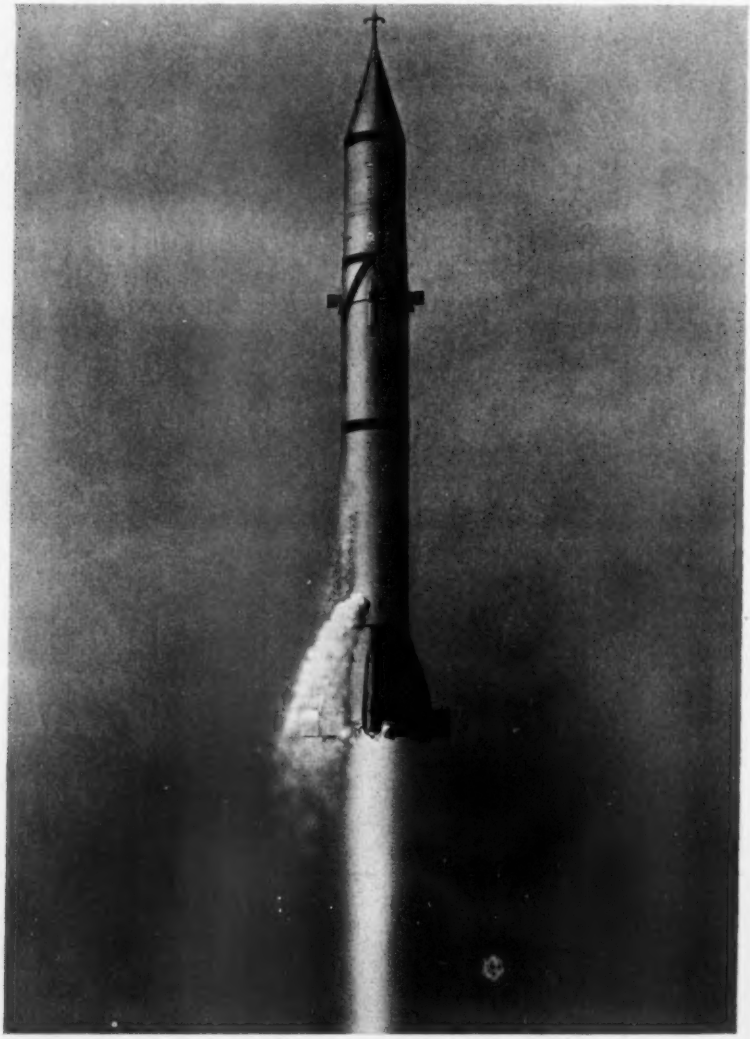
THE BRITISH ASTRONOMICAL ASSOCIATION

Founded 1890, now numbers over 2,600 members. Open to all interested in Astronomy. Chief objects are the association of observers for mutual help, circulation of astronomical information, and encouragement of popular interest in Astronomy. The Association issues a *Journal* about nine times annually, *Circulars* giving current Astronomical news, a *Handbook* annually and *Memoirs* on the work of Sections, including the Sun, Moon, Planets, Comets, Auroræ, and Variable Star Sections.

The Association has a good Lending Library and Lantern Slide Collection. A number of instruments are also available for loan to members.

For further particulars apply to—

The Assistant Secretary, 303, Bath Road, Hounslow West, Middlesex.



Army's Redstone missile in flight at Patrick Air Force Base, Florida.

Compounded Subscription

Members who desire Life Membership may compound their annual subscriptions by the payment of a single lump sum in accordance with the following scale. By compounding subscriptions, members help the Society at a very important period in our history when it is necessary to accumulate capital funds as quickly as possible in order to safeguard our future development, e.g., by the acquisition of alternative premises when our present lease expires, purchase of additional equipment, etc.

Age limits	Membership	Fellowship and Senior membership
	£	£
20–25	36	45
26–30	33	40
31–35	30	36
36–40	27	32
41–45	24	29
46–50	22	26
51–55	20	23
over 55	18	21

It will be noted that the above scale details the rates for compounded subscription payable by Senior Members, these having recently been approved by the Council who decided that the rates should be equivalent to Fellowship in all respects. However, members who apply for transfer to Senior Membership and who have already paid the compounded subscription in accordance with the Membership scale may elect to adopt either of the following:—

(*a*) Remit £1 1s. per annum for the Senior Membership subscription i.e., the difference between the normal Fellowship and Membership subscriptions.

(*b*) Remit the difference between Senior Membership and Membership fees in accordance with the above scale.

Applications may be made at any time and not necessarily on January 1, when the normal subscriptions fall due. Futher enquiries may be addressed to the B.I.S. Secretary if desired.

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SKY PUBLISHING CORPORATION, Harvard Observatory, Cambridge 38, Mass., U.S.A.

ABSTRACTS

Edited by J. HUMPHRIES.

A further addendum to the list of abbreviations of titles of journals included with the 1954 index is given below.

Aviation Ind. Comm.	Aviation—Industrie et Commerce.
J. Camb. Univ. Interplanet. Soc.	Journal of the Cambridge University Interplanetary Society.

AERODYNAMICS

(196) **The heat barrier.** A. F. Pexton. *J. Camb. Univ. Interplanet. Soc.*, **1**, 3–12 (*June*, 1955). Properties of metals at high temperatures, and their application to problems raised by aerodynamic heating of rocket structures. (3 *refs.*)

AIRCRAFT

See also abs. nos. 233, 240 and 243.

(197) **Automatic safety devices for rocket-propelled aircraft.** F. Florio. *Jet Propulsion*, **25**, 235–236 (*May*, 1955). Describes elements closing an electric circuit in the presence of propellants or an excessive temperature.

ASTRONAUTICS

(198) **The Fifth International Astronautical Congress.** A. Eula. *Aerotecnica*, **34**, 278–285 (*Oct.*, 1954) (*in Italian*). Review of proceedings and abstracts of technical papers.

(199) **Problems of the manned rocket.** P. Bradshaw. *J. Camb. Univ. Interplanet Soc.*, **1**, 27–29 (*June*, 1955). On the protection of human beings in space.

(200) **Elements of space-suit design.** M. R. H. Jeffries. *J. Camb. Univ. Interplanet Soc.*, **1**, 25–27 (*June*, 1955). Discusses the requirements to be met by a space-suit from the medical point of view. (1 ref.)

(201) **Colonising the solar system.** D. G. Noel. *J. Camb. Univ. Interplanet. Soc.*, **1**, 51–53 (*June*, 1955). The requirements for successfully establishing human colonies, and the possibility of establishing them on the planets.

(202) **An assessment of the satellite project.** R. A. Smith. *J. Camb. Univ. Interplanet Soc.*, **1**, 19–24 (*June*, 1955). A critical review of the current possibility of a satellite vehicle.

(203) **Roundtable conference on "The Vanguard."** *Aero. Dig.*, **71** (6), 28–30 (*Dec.*, 1955). Report of discussion by M. W. Rosen, F. Zwicky, W. G. Purdy and A. P. Adamson, sponsored by the Amer. Rocket Soc. Applications, orbits and other details of proposed artificial satellites.

(204) **The satellite vehicle for communications and navigation.** H. E. Canney and F. I. Ordway. *Aero. Dig.*, **71** (6), 40, 42, 44, 46 (*Dec.*, 1955). Reviews published work on the uses of the satellite as radio relay for terrestrial transmissions and for transmissions to the Moon, planets and spaceships. (7 *refs.*)

(205) **Scientists reveal "Vanguard" design details.** *Aviation Wk.*, **64** (13), 33–34 (*March* 26, 1956). Further information, especially on method of launching and instrumentation. Three satellite configurations are being considered, one being a minimum weight, one relying on telemetering, one a large inflatable satellite for optical tracking and one a compromise between the two extremes. It is hoped that radio amateurs will be able to provide supplementary data.

(206) **The artificial earth satellite.** S. F. Singer. *Discovery*, **17**, 140–145 (*April*, 1956). Review of applications and methods of obtaining information.

BIOLOGY AND MEDICINE

See also abs. nos. 199 and 200.

(207) **Review of biological effects of subgravity and weightlessness.** D. G. Simons. *Jet Propulsion*, **25**, 209–211 (*May*, 1955). Discusses the effects of sub-gravity and

weightlessness on the organs of balance, sight and sense of position and consequently on orientation and co-ordination of movement. Consequently illusions and sensory inconsistencies, incapacitating the subject under certain conditions. Experiments on man and animals are quoted. (16 *refs.*)

(208) **Planetary ecology.** C. R. Armstrong. *J. Camb. Univ. Interplanet. Soc.*, **1**, 44–45 (*June*, 1955). A discussion of surface conditions on the minor planets. (3 *refs.*)

MATERIALS

(209) **Investigation of carbon (graphite) base materials suitable for rocket and ramjet applications.** W. S. Smiley. *Stanford Res. Inst. Bi-monthly Progr. Rept. No.* 6, 6 pp. (*Feb.-March*, 1954).

(210) **Cermets—new high-temperature materials.** R. Steinitz. *Jet Propulsion*, **25**, 326–330 (*July*, 1955). Describes materials made by combination of metals and refractory compounds by the techniques of powder metallurgy. Compares their properties with those of high-temperature metal alloys and discusses their fields of present and possible future application. (15 *refs.*)

MISCELLANEOUS

(211) **Kazimierz Siemienowicz — Polish pioneer of rocket technology.** S. Probulski. *Technika Lotnicza*, **10**, 66–73 (*May-June*, 1955) (*in Polish*). Review of Siemienowicz's work based mainly on his book, *Artis magnae artilleriae pars prima*. (8 *refs.*)

(212) **Down under at Woomera.** *Jet Propulsion*, **25**, 240 (*May*, 1956). The Australian rocket range briefly described.

PHYSICS

(213) **The reduction of data from the rocket-grenade experiment.** W. G. Stroud. *Fort Monmouth Signal Corps Engng. Labs. Tech. Memo. No. M-1570*, 33 pp. (*April*, 1954).

(214) **The role of the adiabatic exponent in the case of chemically variable gas mixtures.** I. Sänger-Bredt. *Z. angew. Math. Phys.*, **6**, 25–66 (*Jan.*, 1955) (*in German*). Deals with flow-problems. The influences of variable specific heats and partial pressures on the adiabatic exponent are taken separately into account. With new, appropriately determined exponents generalized equations for isentropic flow are established. (7 *refs.*)

PROJECTILES

(215) **R.T.V.-N-12a "Viking."** *Glenn L. Martin Co. Progress Rept. No.* 36, *Eng. Rept. No.* 6373, 17 pp. (*Dec.* 15, 1953–*March* 15, 1954).

(216) **U.S.A.F. "Aerobee" Nos. 38, 39, 40, 41, 42, 43.** E. E. Rasmussen. *Holloman Air Force Base, Holloman Air Devel. Center Final Field Test Repts Nos. HADC-TR-54-9*, 10 11, 12, 13, 14 (*April*, 1954).

(217) **Propellant level sensors for "Viking."** A. W. Niles and R. L. Easton. *Rocket Res. Rept. No.* 17, *Naval Res. Lab. Rept. No.* 4454, 29 pp. (*Dec.* 17, 1954).

(218) **Foreign rockets and guided missiles, 1954.** N. Bowman. *J. Space Flight*, **7** (4), 1–6 (*April*, 1955). Tabulated data. (4 *refs.*)

(219) **Guided missile control; a synoptic review of methods and requirements.** *Bristol Quarterly*, **1**, 157–160 (*Spring*, 1955.)

(220) **The importance of mixture ratio control for large rocket vehicles.** R. H. Reichel. *Jet Propulsion*, **25**, 291–293 (*June*, 1955). For highest performance all fuel and oxidant must be used up completely. Means for achieving this are described. (3 *refs.*)

(221) **Reliability in guided missiles.** R. P. Haviland. *Jet Propulsion*, **25**, 321–325, 330 (*July*, 1955). Results of tests of several missiles are examined to show the nature of the problem of reliability. The theory of reliability is reviewed and examples of life parameters given. The data needed and the steps involved in a satisfactory immediate solution are outlined. (7 *refs.*)

(222) **The world's guided weapons.** A. R. Weyl. *Aeronautics*, **32** (6), 28–38 (*July*, 1955). A comprehensive review of world progress in missile development, including American, Russian, French and British types, and the German missiles from which they have been developed.

(223) **Prediction of missile reliability.** M. J. Kirby and H. R. Powell. *Sperry Engineering Review*, 12–20 (*July-Aug.*, 1955). Development of a testing-to-failure quantitative method for components based on the probability concept of the relationship between fatigue and failure in terms of ultimate strength data.

(224) **Soviet missile progress.** A. J. Zaehringer. *Aero. Dig.*, **71** (6), 48, 50–51 (*Dec.*, 1955). Graphical presentation and discussion of probable Soviet missile types.

(225) **Army's "Redstone".** *Aviation Wk.*, **64** (9), 31 (*Feb.* 27, 1956). Photos of static firing and take-off of Redstone missile.

(226) **U.S. missiles and pilotless aircraft.** *Aviation Wk.*, **64** (11), 217 (*March* 12, 1956). Table of specifications of current projects.

(227) **New "Scorpion" designed for Falcon.** *Aviation Wk.*, **64** (12), 32–33 (*March* 19, 1956). Action photos.

(228) **Soviet advances force U.S. missile pace.** D. A. Anderton. *Aviation Wk.*, **64** (11), 98–101, 103 (*March* 12, 1956). Review of types of missiles and rocket propellants.

(229) **Effective I.C.B.M. defense possible under present state of art.** D. A. Anderton. *Aviation Wk.*, **64** (15), 44–46 (*April* 9, 1956). With an I.C.B.M. of 5,000 miles range an anti-missile missile would have an allowable time of 6 to 7 secs. to effect a kill. Various factors involved are analysed.

(230) **Is missile reliability worth the cost?** I. Stone. *Aviation Wk.*, **64** (16), 50–52 (*April* 16, 1956). Compares the increased cost of improving the reliability with the savings in the numbers of missiles and backing-up equipment.

(231) **Some principles of missile guidance.** *Aeroplane*, **90**, 280–281 (*April* 20, 1956). Review of systems for surface-to-air, air-to-air and ballistic missiles.

RADIO AND ELECTRONICS

See also abs. no. 204.

(232) **Telemetering.** A. A. McKenzie and H. A. Manoogian. *Electronics*, **29** (4), 153–180 (*April*, 1956). A series of short articles on systems development, input transducers, signalling methods, commutating devices and output indicators. A glossary and comprehensive bibliography are included.

ROCKET MOTORS

(233) **Notes on development of J.A.T.O.** H. D. Greenburg. *Picatinny Arsenal, Ordnance Res. Dev. Div. P.A. Notes No.* 62, 16 pp. (*June*, 1953).

(234) **Similarity analysis for chemical reactors and the sealing of liquid-fuel rocket engines.** S. S. Penner. *Calif. Inst. Tech., Jet Prop. Center Tech. Rept. No.* 8, 41 pp. (*Nov.*, 1954).

(235) **Combustion instability in liquid-propellant rocket motors.** *Quart. Progr. Rept., Aug.* 1–*Oct.* 31, 1954. *Princeton Univ. Dept. Aero. Engng. Rept. No.* 216–*J*, 39 pp. (*Dec.*, 1954).

(236) **Combustion instability in liquid-propellant rocket motors. Low-frequency oscillatory phenomena.** E. Maciore. *Aerotecnica*, **34**, 319–325 (*Dec.*, 1954) (*in Italian*). General consideration of the various methods of analysis of a system with constant pressure propellant feed. (8 *refs.*)

(237) **Mixture ratio and temperature surveys of ammonia-oxygen rocket motor combustion chambers.** D. I. Baker. *Jet ropulsion*, **25**, 217–236 (*May*, 1955). Describes an investigation of combustion temperatures and gas composition at many points within the combustion chamber by the use of water-cooled probes. Two injector patterns and a range of mixture ratios were used. Theoretical calculations for the ammonia-oxygen system are included. (14 *refs.*)

(238) **Design of rocket thrust chambers.** E. E. Dawson. *J. Camb. Univ. Interplanet. Soc.*, **1**, 15–18 (*June*, 1955).

(239) **Testing of liquid-propellant rocket engines.** W. N. Neat. *J. Camb. Univ. Interplanet. Soc.*, **1**, 13–15 (*June*, 1955). Describes the facilities required for testing rocket motors and outlines the procedure used in development of a motor.

(240) **The de Havilland "Spectre."** *D.H. Gaz.*, **87**, 56 (*July*, 1955). Information on the purposes and application of a new liquid-propellant rocket engine.

(241) **Static seals for missile applications.** R. R. Ashmead. *Jet Propulsion*, **25**, 331–333, 340, 346 (*July*, 1955). Discusses sealing over range — 300 to + 1,400° F., suitable plastic and metallic seals and their physical properties, and the problems of seal design. (2 *refs.*)

(242) **Static seal for low-temperature.** S. E. Logan. *Jet Propulsion*, **25**, 334–340 (*July*, 1955). Describes the theory and practice of sealing low-temperature fluids such as liquid oxygen at moderately high pressures. Gives details of the use of differential expansion to load neoprene sealing surfaces. (15 *refs.*)

(243) **Superperformance rockets for fighter aircraft.** R. Gordon. *Aero. Engng. Rev.*, **14** (8), 27–28, 35 (*Aug.*, 1955). A proposal that designs of all future air frames provide for possible rocket installations and that rocket subassemblies may thereby become standardized. (7 *refs.*)

(244) **Apparatus for feeding a liquid fuel, a liquid oxidizer and water to a combustion chamber associated with rocket apparatus.** R. H. Goddard. *U.S. Pat. No.* 2,722,100 (*Nov.* 1, 1955). Injector with target plate.

(245) **Controlling ballistic missile powerplants.** R. H. Reichel. *Aero Dig.*, **71** (6), 22–27 (*Dec.*, 1955). Various methods of controlling thrust and mixture ratio both during Static firing and flight are considered. Effect of missile acceleration is discussed. (10 *refs.*)

(246) **Improvements in or relating to the manufacture of double-walled tubular articles of varying cross-section.** W. A. Schoenheit. *Brit. Pat. No.* 742,390 (*Dec.* 30, 1955). A method of making regeneratively cooled combustion chambers by hydro-forming individual tubes and welding together a multiplicity of the formed tubes.

(247) **Modern techniques in solid rocket engineering.** G. S. Sutherland. *Aero. Dig.*, **72** (1), 46–48, 51–52, 54, 56 (*Jan.*, 1956). Reviews existing propellants, both double-base and composite and discusses charge design, casing design and ignition. An example of charge design is given. (29 *refs.*)

(248) **Aerojet builds new missile rocket plant.** *Aviation Wk.*, **64** (12), 37, 39–40 (*March* 19, 1956). This new site houses both solid and liquid propellant facilities and has test beds capable of testing motors up to 1.5 million lb. thrust.

(249) **Improvements in and relating to starting valves for rocket motors.** Oerlikon Buehrle & Co. *Brit. Pat. No.* 747,139 (*March* 28, 1956). Slow-opening valve operated by propellant pressure and controlled by orifice.

(250) **Atlas engine is fired by Rocketdyne.** R. Sweeney. *Aviation Wk.*, **64** (17), 37–38 (*April* 23, 1956). Description of North American Aviation's rocket test facilities.

ROCKET PROPELLANTS

See also abs. nos. 217 and 237.

(251) **Liquid rocket fuels.** M. A. Pino. *Calif. Res. Corp. Final Rept.*, 106 pp. (*Oct.*, 1951–*June*, 1953).

(252) **Research on ultra-energy fuels for rocket propulsion.** C. L. Randolph. *Aerojet-General Corp. Rept. No.* 1219–1, 7 pp. (*June* 18–*Sept.* 17, 1954).

(253) **Ignition-delay determinations of furfuryl alcohol and mixed butyl mercaptans with various white fuming nitric acids using modified open-cup and small-scale rocket engine apparatus.** D. J. Ladanyi, R. O. Miller & G. Hennings. *N.A.C.A. R.M. E.53E.29*, 17 pp. (*Feb.*, 1955).

ROCKET PROPULSION

(254) **The future of rocket propulsion.** A. D. Baxter. *J. Camb. Univ. Interplanet. Soc.*, **1**, 29–40 (*June*, 1955). A statement of the problems of structural design, propellants, and the application of nuclear energy to the spaceships of the future.

(255) **Rockets through space and time.** J. Eloy. *Aviation Ind. Comm.*, **3** (6), 9, 32 (*June*, 1955) (*in French*). Historical account of rockets and rocket weapon development.

(256) **The problem of propulsion.** M. W. Windsor. *J. Camb. Univ. Interplanet. Soc.*, **1**, 40–44 (*June*, 1955). The application of rocket propulsion to interplanetary travel.

(257) **The incremental-step rocket in free space.** H. R. Wahlin. *Jet Propulsion*, **25**, 347–349 (*July*, 1955). Presents the theory of the incremental-step rocket in an easily manageable form, and gives their advantages and limitations. (2 *refs.*)

REVIEW

The Planet Venus

(By Patrick Moore. 132 pp. with 8 plates and 13 figures. Faber and Faber, Ltd., London, 1956. 15s.)

It is perhaps misleading to say that this book contains 132 pages. If we omit title pages and contents pages at the beginning, references and index at the end but include the appendices, there are 112 pages, a slimness in content which makes the price seem rather high. It might have been better if this book had been published at one-third the price in smaller form with paper covers.

The author describes what is known about the planet Venus and assembles some of the conflicting theories that have been put forward to explain the observed facts which, as any astronomer knows, are lamentably few due to the impenetrable atmosphere surrounding this planet.

It is difficult to know what class of readers the author aims at—certainly not the professional astronomer for the known facts about Venus are readily available to him, also the more recent theories of the planet's constitution by Whipple, Menzel and Wildt. Other readers may find the subject-matter of the book interesting but the treatment of it is spoiled by the indiscriminate inclusion of outdated and absurd theories, errors that should have been removed before publication and woolly and misleading phraseology.

For example on page 25 the author states that the presence in large quantities of carbon dioxide in Venus's atmosphere is probably the greatest mystery of the Solar System. One can think of many mysteries to rival that problem in perplexity.

On page 48 the Earth's axis is inclined to the plane of its orbit at an angle of $23\frac{1}{2}°$, a figure that should be $66\frac{1}{2}°$. This error is repeated with the other planets.

On page 49 Uranus is surely not so odd as the author implies. The idea that "Uranus rolls along almost pole-first in its journey round the Sun" suggests that the planet's axis of rotation itself rotates in the period of revolution of Uranus, which, needless to say, is not so.

In chapter 7 is it really necessary to give in detail all the old estimates of the period of rotation of Venus when we now know how unreal they were?

On page 51, the author makes the statement that the surface of Venus is "either very warm or very hot." We are left to fit temperatures to these two classifications.

On page 59, figure 12 should refer surely to plate 8b which in any case is surely superfluous, a blinding glimpse into the obvious.

On page 76 we are told that carbon dioxide is "the choking gas that we see dissolved in soda-water," a contradiction in terms.

On page 99 the suggestion is made that a couple of triangles whose sides are thousands of times as long as their bases should be drawn to scale in order to work out the Sun's distance.

On page 111 the giant planets are said to "have gaseous surfaces covered with dense, poisonous atmospheres," a confusion of the visible surface with the possibly solid surface of the core.

The author is probably too dogmatic on page 112 about the certainty that all forms of life in the universe are fundamentally, though not perhaps superficially, similar to life forms found on the Earth. Even on Earth some pretty "alien" life forms can be found such as the bacillus Boracicola which can exist happily in a saturated solution of boric acid, to quote one example.

This book can certainly not be said to be, in the words of the publisher, a necessary addition to the library of any astronomically minded person and it is unfortunate that the author chose as his subject a planet which even today remains almost a complete mystery.

A. E. Roy.

Mr. Moore has commented on the above Review as follows:—

"It is a pity that Dr. A. E. Roy has made so many elementary slips in his review of my book. A few should be pointed out, in the interests of accuracy.

"Roy refers to 'the more recent theories of the planet's constitution by Whipple, Menzel and Wildt.' Whipple and Menzel have produced their theory of the surface and atmospheric conditions in collaboration; Wildt's was produced some time ago. Roy's statement is therefore confusing.

"Roy says that I make the statement that the surface of Venus is 'either very warm or very hot': 'we are left to fit temperatures to these two classifications.' Quite so. The actual temperature is not known with precision; 'hot' indicates greater temperature than 'warm,' as can be verified by reference to an English dictionary.

"Roy asks whether 'it is necessary to give in detail all the old estimates of the period of rotation of Venus when we now know how unreal they were.' In a serious review, this is a most curious question. Recent estimates of the rotation period range from 224¾ days, the length of the sidereal period (Dollfus) down to only a few days; we simply do not know. It is therefore impossible to dismiss any estimate as 'unreal,' and had Roy studied the relevant part of my book he would have appreciated that we cannot summarily dismiss many of the old observations as spurious, even though we may have some doubts. All but the first four of the references I have listed are comparatively modern, and are thus worthy of close attention; it is evident that Roy has not, for instance, heard of the series of fine drawings made by Trouvelot during the last part of the nineteenth century. His comment is therefore extremely misleading, and he should surely have taken more care before writing it.

"Roy says that on page 99 'the suggestion is made that a couple of triangles whose sides are thousands of times as long as their bases should be drawn to scale in order to work out the Sun's distance.' Actually, page 99 deals almost entirely with Horrocks' visual observations of the transit of Venus of 1639. I can only assume that Roy has carelessly written '99' instead of 'page 100,' but this makes no difference to the value of his comment, since I have made no such suggestion. I pointed out, quite correctly, that Halley had shown how transits could be used to

measure the Sun's distance, and then I went on: 'In practice, the procedure followed is naturally more complicated, and instead of measuring the exact position of Venus against the solar disc it was found better merely to measure the moment when the planet passed fully on to the Sun. As the whole method is now completely obsolete, there is no point in describing it further.'

"Such were my words. Yet Roy states that I suggest 'drawing a couple of triangles whose sides are thousands of times as long as their bases,' and, moreover he quotes the wrong page. I stress this particular mistake on his part because it is a key to the whole standard of his criticisms.

"On Roy's reference to 'the possibly solid surface of the core' of a giant planet, it must be pointed out that our knowledge of the constitution of the giant planets is still very limited, and Roy's statement is likely to mislead the non-specialist into thinking that our information is reliable.

"The comments made on my page 112 *are, I fear, somewhat muddled. Roy points out, quite correctly, that remarkable life forms can exist on Earth, and this I have never questioned; but this is completely irrelevant to my argument as put forward in the book, as should be evident to anyone who has read it.*

"For the rest, Roy is right in pointing out two proof slips, both very minor, both made quite clear by the diagrams, and neither of which is relevant to his other comments. His reference to the 'indiscriminate inclusion of outdated and absurd theories' is amusing; I should be interested to know which parts of the book he means. The puerile humour (I assume it to be humour) of 'a blinding glimpse into the obvious' is best ignored.

"There are many other mistakes of fact and quotation in the review, but it would be pointless to discuss them all, and I have no wish to embarrass Dr. Roy more than I must. Nor is it advisable to discuss phrases such as 'a blinding glimpse of the obvious,' which are not in my view suitable for publication in a serious journal. I regret the necessity of replying at all; but in this age of specialization, it is essential to prevent the non-specialist from being misled."

DR. ROY has provided the following remarks on the comments made by Mr. Moore:—

"It is doubtless not unusual for an author to disagree with his reviewers, but it is surely rare for an author publicly to display his personal resentment against the review. Since Mr. Moore has chosen to act in this way, as the reviewer I have no alternative but to enlarge on the sections to which Mr. Moore has taken exception. I will ignore the general and completely unsubstantiated statements about 'many elementary slips' and 'many other mistakes of fact and quotation' (which latter are presumably supposed to occur in the small residuum of my review not dealt with in detail) and confine myself to the specific objections of Mr. Moore which he takes to indicate the carelessness of my review.

"He attaches great importance to the fact that I have given as page 99, the reference to the determination of the Sun's distance by transit of Venus. Mr. Moore considers that this should be page 100, but if the reader refers to the book he will find that the paragraph and argument in which the statement

in question occurs begin on page 99. Furthermore, Mr. Moore denies making the suggestion referred to in my review. I am therefore at a loss to understand the meaning of the following paragraph which begins at the foot of page 99:—

> "'Before the 1761 transit took place, Edmond Halley, following up an earlier suggestion by James Gregory, realized that these passages of Venus might be used to measure the distance of the Sun from the Earth. Fig. 12 shows the situation during a transit. To an observer at A, Venus will be seen in position A′: to an observer at B, Venus will appear at B′. If simultaneous observations are made from A and B, and the distance AB is known, *the entire diagram can be drawn to scale*, and the distance of the Sun worked out.' (The italics are mine.)

"Mr. Moore's subsequent qualification of this statement is in my opinion wholly inadequate.

"Mr. Moore considers that the mis-statement of *six* angles, page 48, and the reference to Plate 8*b* as Fig. 12, constitute two proof slips, both very minor, but regards the 'mistake' of one page reference in my review (99 for 100) as a key to the whole standard of my criticisms.

"In this light, let us look a little more closely at the 'very minor proof slip' on page 48. Mr. Moore states: 'The Earth's axis is inclined to the plane of its orbit at an angle of 23½°.' This statement should read: 'The Earth's axis is inclined to the normal to the plane of its orbit at an angle of about 23½°.' He then continues: 'Most of the other planets have similarly tilted axes,' and gives figures for these inclinations that are approximately 90° minus the correct figures. But it should be further noted that even after Mr. Moore's definition has been corrected, the figures given in his text are in some cases in disagreement not only with the true values, but also with the corresponding figures given in his diagram appearing on the *same* page. The following table shows the discrepancies.

Planet	Moore diagram page 48	Moore text page 48	True value*
Earth	23° 45′	23½°	23° 27′
Mars	25° 2′	25½°	25° 12′
Jupiter	3° 1′	3°	3° 7′
Saturn	26° 75′ (*sic.*)	26¾°	26° 45′
Uranus	98°	98°	97° 59′
Neptune	29°	29°	29°

* The figures given in the last column are the inclination of the axis of rotation to the normal to the planet's orbit and are taken from Allen, *Astrophysical Quantities*, The Athlone Press, 1955, corrected to the nearest minute of arc, where possible.

"I find it difficult to understand why Mr. Moore should find confusing my statement regarding the more recent theories of Whipple, Menzel and Wildt—all published within the last thirty years. Mr. Moore's history covers more than two hundred years.

"The fact that, as Mr. Moore points out, modern determinations of the rotation period of Venus are still inconsistent, in no way invalidates my

contention that no weight can be attached to the old estimates of this period, nor my opinion that their detailed discussion in this book is superfluous. Even should the finally accepted figure agree approximately with one of these earlier determinations, one would not be justified in attaching any significance to the latter.

"Mr. Moore takes me to task for my reference to the possibly solid surface of the core of a giant planet, yet he states on page 111 that '. . . the giant planets are not only bitterly chill but have gaseous surfaces covered with dense, poisonous atmospheres.' Surely the charge of misleading the non-specialist into thinking our information is reliable is misdirected since his statement exhibits the dogmatism of which he is accusing me.

"I am still unrepentant regarding my comments on Mr. Moore's Earth-bound views on the nature of life and am confident that the reader of Mr. Moore's book will appreciate my viewpoint. All the conditions he lays down on page 112 for the existence of 'creatures' are based upon our terrestrial experience, so that he is adopting the mediaeval attitude that the Earth is the measure of the Universe.

"The phrase, 'a blinding glimpse into the obvious,' may be puerile; it is not more so than the diagram it describes. I cannot conceive of a more obvious waste of money than the reproduction of so crude an illustration of the rotation of a sphere (purporting, without justification, to illustrate the Doppler effect) as a half-tone plate, when a simple line diagram would have sufficed.

"I am well aware that the temperature at the surface of Venus is not known with precision, and I am also aware that the term 'hot' is usually assumed to denote a higher temperature than 'warm.' But does Mr. Moore seriously consider that a qualitative scale—warm, hot, very warm, very hot—in this order, is a meaningful and not misleading scale to describe planetary temperatures (and where does 'bitterly chill' fit in)?

"Mr. Moore does not wish the non-specialist to be misled. I am in entire agreement; this is the very reason why I wrote this review. In conclusion, may I reassure Mr. Moore that his comments have not caused me a moment's embarrassment."

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In view of the fact that a List of Members can be issued only at infrequent intervals, the Council has decided to provide information on the elections made at each meeting, as it is felt that this is likely to prove of interest to other members.

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